AFRPL TR-72-45

FINAL REPORT

ORBIT-TO-ORBIT SHUTTLE ENGINE DESIGN STUDY

Contract F04611-71-C-0040

<u>BOOK 3</u>

W. P. Luscher, et. al.
Aerojet Liquid Rocket Company
Sacramento, California

May 1972

Air Force Rocket Propulsion Laboratory
Edwards Air Force Base, California

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Headquarters, Air Force Flight Test Center
Rocket Propulsion Laboratory
Edwards Air Force Base, California

FOREWORD

This report presents the work accomplished on Contract F04611-71-C004, the Orbit-to-Orbit Shuttle Engine Design Study (OOS) over the period from 1 March 71 to 1 December 1971. The program was admistered by the Procurement Division of the Directorate of Material, Edwards Air Force Base, Edwards, California. The technical project manager at the Rocket Propulsion Laboratory, Edwards, California was Mr. L. Tepe. Mr. Werner P. Luscher directed the study effort for Aerojet Liquid Rocket Company.

This report is contained in 4 books described as follows:

Book 1: Parametric Cycle Study

Book 2: 25K lb Engine Design

Book 3: 25K lb Engine Maintenance, Development Plans, Cost Estimates and 10K lb Engine Design

Book 4: Appendices

This technical report has been reviewed and is approved.

L. E. Tepe
Project Manager

ABSTRACT

This report presents the analytical design of propulsion systems utilizing LOX/Hydrogen propellants to be used as the propulsion for the Orbit to Orbit Space Vehicle of 65,000 lb lift-off weight.

The report contains the evaluation of various engine cycles in the thrust range of 8,000 lb to 50,000 lb thrust for performance, weight and envelope culminating in the cycle selection and detail design of a 25,000 lb and 10,000 lb thrust engine. The engine concepts are described in sufficient detail to obtain reliable engine weight, performance, envelope information and methods of engine control. The impact of various engine design requirements were evaluated. The engines are designed to be reusable and capable of starting in the idle mode operation.

The technology requirements for meeting the engine design and operating requirements are identified.

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NOMENCLATURE

UNITS:

°F	Degrees Farenheit
ft	Feet
GPM	Gallons per Minute
hr	Hours
HP	Horsepower
in.	Inches
lb	Pounds
mm	Millimeters
psi	Pounds per Square Inch
rpm	Revolutions per Minute
°R	Degrees Rankine
sec	Seconds

SYMBOLS:

A/R	Area Ratio
CR	Contraction Ratio
DN	Bearing (Bore Diameter mm x rpm)
ε	Area Ratio
F	Thrust
I_s	Specific Impulse
L'	Chamber Length
MR	Mixture Ratio
M	Mach Number
MRD	Mixture Ratio Distribution
Ns	Specific Speed
N_f	Number of Cycles
P_c	Chamber Pressure
PVC	Pressure Volume Compensated
SF	Safety Factor
S	Suction Specific Speed
T_{T_i}	Turbine Inlet Temp
T_{FD}	Fuel Pump Discharge Temp
T	Chamber Hot Wall Temp
T_w	Chamber Wall Temp Gradient
T_B	Bulk Temperature
V	Velocity
$\dot{w}$	Weight Flow Rate
W_{BO}	Burnout Weight
η	Efficiency

GLOSSARY:

ALRC	Aerojet Liquid Rocket Company
AGCarb	Carbon Cloth Material
CJKT	Preburner & Turbine By-pass
ERE	Energy Release Efficiency
FTPA	Fuel Turbopump Assembly
FDV	Fuel Discharge Valve
FRHG	Fuel Rich Hot Gas
FPBYV	Fuel Preburner & Turbine By-Pass Valve
FFC	Final Flight Configuration
Hz	Frequency
LRU	Line Replaceable Unit
N_{TF}	Fuel Turbine rpm
N_{TO}	Oxidizer Turbine rpm
N_{TFE}	Fuel Low Speed Inducer rpm
N_{TOE}	Ox Low Speed Inducer rpm
NPSH	Net Position Suction Head
NPSP	Net Positive Suction Pressure
OTPA	Oxidizer Turbopump Assembly
OPBV	Oxidizer Preburner Valve
OSCV	Oxidizer Mixture Ratio Control Valve
PL	Payload
PFC	Preliminary Flight Configuration
TCA	Thrust Chamber Assembly
TGJD	Temp at FRHG Injector Inlet
UTMO	Max Oxidizer Turbine Tip Speed
UTMF	Max Fuel Turbine Tip Speed

C. ENGINE MAINTENANCE (25K ENGINE DESIGN)

1. Ground-Based Maintenance

Engine maintenance was a basic requirement in the overall engine design as well as the subassembly and component designs. Because the purpose of this study was to provide a preliminary engine design and engine system data, the maintainability effort was directed toward the impact of maintainability on engine design both from the standpoint of instrumentation and maintainability design features rather than definition of ground support equipment and logistics problems. The maintenance program followed a step-by-step evolution from establishment of maintenance concepts, performance of failure modes and effects analysis, definition of line-replaceable units and incorporation of maintainability features. The following paragraphs present the step-by-step procedure followed in assuring that the OOS engine design contains maintainability features.

a. Maintenance Concepts

The maintenance concepts or philosophy establishes the basis for all maintainability decisions and the ground rules followed in deriving the maintainability program. For the OOS program the following maintenance concepts were used:

Maximum utilization of the onboard engine controller and instrumentation for engine checkout, engine monitoring during flight for maintenance significant trends, and fault isolation during flight.

A leave-it-alone-if-working philosophy*.

On-the-vehicle maintenance (ground-based) by removal and replacement of Line Replacable Units (LRU).

The OOS System achieves a reduction of the payload-in-orbit-cost by reusability of the delivery system. Reusability requires that the system be maintained to obtain reliable payload delivery of the same system in subsequent missions.

The method of maintenance is largely dependent upon the availability of payloads and fleet size which determine the OOS stage turn around time. The turn around time effects the methods by which flight readiness of the stage will be established. The basis of flight readiness is the engine performance of the last flight recorded by the onboard instrumentation. Reduction and analysis of this flight data is the basis from which maintenance decisions are made. Several methods of data retrieval are feasible and the selection is based on the vehicle turn around time.

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*This does not include those components designed to give a certain minimum life capability to achieve high performance or low weight.

Real time data retrieved by telemetry via communication satelite.

Vehicle in-flight data recording and ground playback.

Vehicle data recording and in-flight reduction and analysis.

The method used will change during the flight program as the reliability of the system is increasing. At the beginning of the flight, real time data is required since the system is still in development. As reliability increases, more and more data storage will be used. The maintenance concept and turn around time will also change during the flight program and no final decision on data retrieval and instrumentation requirement can be made at this time.

Engine maintenance costs are largely dependent on the component life capability. Maintenance costs are therefore designed into the engine by the stated component design life goals (300 cycles, 10 hours) and by the ease of engine inspection and failure detection.

Ground Maintenance Approach

There are three different levels of maintenance considered:

Routine maintenance
Refurbishing
Engine overhaul

The engine design will incorporate specific requirements for each of these levels. The maintenance levels performed between missions are dependent on the progressive system reliability history and will change as the flight program progresses.

Routine Maintenance

This maintenance will be performed with the engine installed on the vehicle and considers the following operations:

Visual inspection of the preburner and TCA injector and chamber.

Engine leak check.

Electrical system continuity check.

Engine filter replacement (Bearings).

Spark plug inspection and check.

Instrumentation replacement, if required.

No flow check or functional check will be performed since the previous flight data will indicate system flight readiness.

Engine Refurbishment

In this maintenance, components of known life limitation or performance degradation will be refurbished. It is most likely that this maintenance is not done on the vehicle and the vehicle will receive a different engine for the next flight. Only the component (LRU) of questionable reliability will be replaced on the engine. Prior to installation into system or storage, the refurbished engine will be tested. No engine testing on the vehicle is considered. Therefore, only LRU's which can be functionally checked out without engine firing will be replaced on the system if replacement time is shorter than engine change.

Engine Overhaul

In this maintenance, the engine is completely disassembled to the subcomponent level, inspected, and parts will be replaced. The engine will be reassembled from functional LRU's available. This means, that there is no engine life in a real flight service but LRU's of various life are assembled into engine assembly. This method of maintenance requires serialization of lowest subassembly parts and parts accountability methods.

The overhauled engine will be tested as an assembly prior to installation into vehicle or storage.

Engine Maintenance and Maintenance Cost Design Goals

The maintenance concepts utilizing LRU's do not recognize an engine assembly as such. For the initial engine design, the following goals are stated:

OOS ENGINE DESIGN STUDY - ENGINE SYSTEMS MAINTENANCE AND MAINTENANCE COST GOALS

REQUIREMENTS	SERVICE FREE OPERATION	BETWEEN OVERHAULS	TOTAL SYSTEM LIFE
Number of Thermal Cycles	60	300	1500
Hours Life	2	10	50
Number of Starts	60	300	1500
Maximum Single Run Time, Sec.	1000	1000	1000
Maintenance Cost/Initial Cost, %	5	25	200

If the engine is completely new, then a maintenance plan and cost estimate can be established ideally as follows:

MAINTENANCE CYCLE PLAN BETWEEN OVERHAULS

MAINTENANCE	ACTIVITY	ESTIMATED COST, PERCENT OF ENGINE	TIMING
Refurbishment 1	Inspection Only	1.0	2 hours or 60 cycles
Refurbishment 2	Inspection Only	1.0	4 hours or 120 cycles
Refurbishment 3	REFURBISH OTPA and Inspection	4.5	6 hours or 180 cycles
Refurbishment 4	REFURBISH FTPA and Inspection	5.0	8 hours or 240 cycles
Overhaul A	Combustion Chamber, Nozzle and Inspection	24.5	10 hours or 300 cycles

Maintenance Engine Design Considerations

The impact of the engine maintenance requirement on the engine design can be defined based on the maintenance concept described.

Provisions have to be made to visually inspect critical components such as the thrust chamber assembly and preburner chamber.

Protective filters for bearing coolant flow have to be designed such as to be easily removable and accessible.

The engine has to be capable of being leak checked in the installed condition.

Instrumentation and sensors have to be replaceable and accessable in the installation (see Page 661).

Replaceable units have to be defined based on life capability and component replacement time.

Flight instrumentation has to be defined to permit definition of engine flight readiness. Instrumentation redundancy or cross check computer has to be employed to assure reliable data.

The engine has to be capable of being ground handled. This applies particularly to nozzle extensions and lines where very thin tube wall thickness should be avoided.

The engine should be capable of being fired for checkout at sea level without impairing the engine operating conditions or structural integrity.

This is probably the most severe design impact and requires definition of available facilities for engine checkout firing.

The large area ratio bell nozzles experience adverse pressure conditions at sea level and atmospheric conditions which tend to collapse the nozzle. Flow separation will occur at certain nozzle pressure ratios which may induce pressure oscillations. Heat transfer conditions in regenerative cooled nozzle extensions will differ from actual in space operating conditions. This may not be of significance in a stage combustion cycle with adaptive thrust and mixture ratio engine control since it can compensate for this fact. Open loop testing would result in a drift of operating conditions. Testing of large area ratio nozzles at sea level requires facilities with steam ejectors. This method would permit demonstration of actual altitude operating mode.

More economical methods would be a facility with an aspirator. This method requires a separable nozzle extension and would not duplicate the altitude operating condition but may be acceptable to demonstrate mechanical and functional integrity of the engine. Separable nozzle extensions considered are dump cooled extensions and radiation cooled extensions. The radiation cooled nozzle extension appears more attractive since it will not complicate engine leak checks.

The final engine design features a fixed all regenerative cooled nozzle and represents the most desirable design but also is most demanding on engine checkout facilities since it requires the availability of an on file high altitude facility.

b. Failure Modes and Effects Analysis (FMEA)

A failure modes and effects analysis was performed for each engine subsystem to determine its mode of failure and the effect on mission objectives, crew safety, and other engine subsystems. Presented in Section III.B.7 is a discussion of the failure modes and effects analysis. The main output of this analysis as it pertains to maintainability is as input data in determining Line Replaceable Units (LRU) by determining failure modes and in determining malfunction detection sensors which isolate the failed component.

c. Component Reliability Assessment

Presented in Table LXXVIII, Section III.B.6 is the reliability apportionment for each of the OOS engine subsystems. This data

is used in establishing which components have a high failure rate and therefore require isolation as an LRU. Additionally, the parts within a component are defined from the reliability assessment and FMEA which require maintainability features which allows for easy replacement.

d. Component Life Cycle Capability

Another factor which influences the selection of LRUs is the life cycle capability of the components. Each of the critical components was evaluated to determine their life capabilities. The results of this evaluation is shown in Table LXXXVIII. Those components which are subject to failure due to cyclic fatigue caused by thermal gradients, such as turbine rotors and thrust chambers, have their life capabilities expressed in cycles. Those components subject to wear or other type failure due to duration, such as bearings, have been expressed in terms of hours.

e. Line Replaceable Units (LRUs)

Presented in Table LXXXIX is a list of the LRUs selected for the OOS engine system. Figure 269 presents an exploded view of the LRUs. Data from the FMEA, the reliability assessment and the component life cycle capability was utilized in deriving the LRUs. Selection of the LRUs was essentially a compromise between reliability/life cycle and ease of removal. For example, even though the turbopump bearings are a life-limited item, removal of the full turbopump assembly is required since no simplified approach for replacing the bearings in place could be derived. All instrumentation has been identified as LRUs because of the relatively low level of reliability and because of the ease of replacement. Interconnecting lines result as LRUs not because of their poor reliability or life cycle capability but because they contain the other half of the flange which connects those items requiring replacement.

f. Incorporation of Maintenance Provisions

Design reviews were held to assure incorporation of maintainability features into the basic designs. Most of the design features are discussed in the sections dealing with design descriptions. Some of the more pertinent maintainability features are discussed in the following paragraphs.

As a general rule, all of the attaching joints will employ bolted flanges in conjunction with K-seals manufactured by Harrison Manufacturing Company. Although bolted flanges are not optimum from a time removal standpoint, their use is justified because of their excellent sealing capabilities due to the even loading imparted to the flange. Flat interfaces which require minimum separation of the flanges for removal have been incorporated in all lines, components and mating surfaces. The selection of the K-seal for all static applications will facilitate interchangeability.

TABLE LXXXVIII

COMPONENT OPERATING CYCLE LIFE CAPABILITY
25K THRUST

1. TPA's

H_2 Turbine Rotors	-	300 cycles
O_2 Turbine Rotors	-	300 cycles
H_2 Turbine Nozzles	-	1500 cycles
O_2 Turbine Nozzles	-	1500 cycles
H_2 Impeller	-	300 cycles
O_2 Impeller	-	700 cycles
H_2 Bearings	-	10 and 19 hr for turbine side bearings
	-	50 hr for pump side bearings
O_2 Bearings	-	150 hr for the turbine side bearings
	-	11 and 164 hr for the pump side bearings
H_2 Shaft Seals	-	1500 cycles
O_2 Shaft Seals	-	1500 cycles
Other Components	-	Exceeds 50 hr/1500 cycles

2. Combustion Components

Preburner	-	1500 cycles
Hot Gas Manifold	-	1500 cycles
Injector	-	1500 cycles
Combustion Chamber	-	300 cycles
Nozzle	-	1500 cycles

3. Valves

All Valves	-	1500 cycles

4. Propellant Lines

Gimballed Lines	-	1500 cycles
Other Propellant Lines	-	Exceeds 1500 cycles

5. Electronics - 50 hours service life

TABLE LXXXIX

LINE REPLACEABLE UNITS (LRU)

Fuel Propellant Circuit

- Fuel Suction Vaned Elbow
- Fuel Turbopump
- Fuel Discharge Valve
- Fuel Discharge Line
- Fuel Start Bypass-Valve
- Fuel Line to Preburner

Oxidizer Propellant Circuit

- Oxidizer Suction Vaned Elbow
- Oxidizer Turbopump
- Oxidizer Discharge Valve
- Oxidizer Discharge Line
- Oxidizer Flow Meter (Discharge Line)
- Oxidizer Line to Preburner
- Oxidizer Flow Meter (Preburner Line)

Combustion Circuit

- Preburner Assembly
- Hot Gas Manifold
- Injector
- Thrust Chamber Assembly (Chamber plus Nozzle)

Electronics

- Instrumentation Transducers
- Instrumentation Harnesses
- Controller
- Control Harnesses

Others

- Gimbal Block

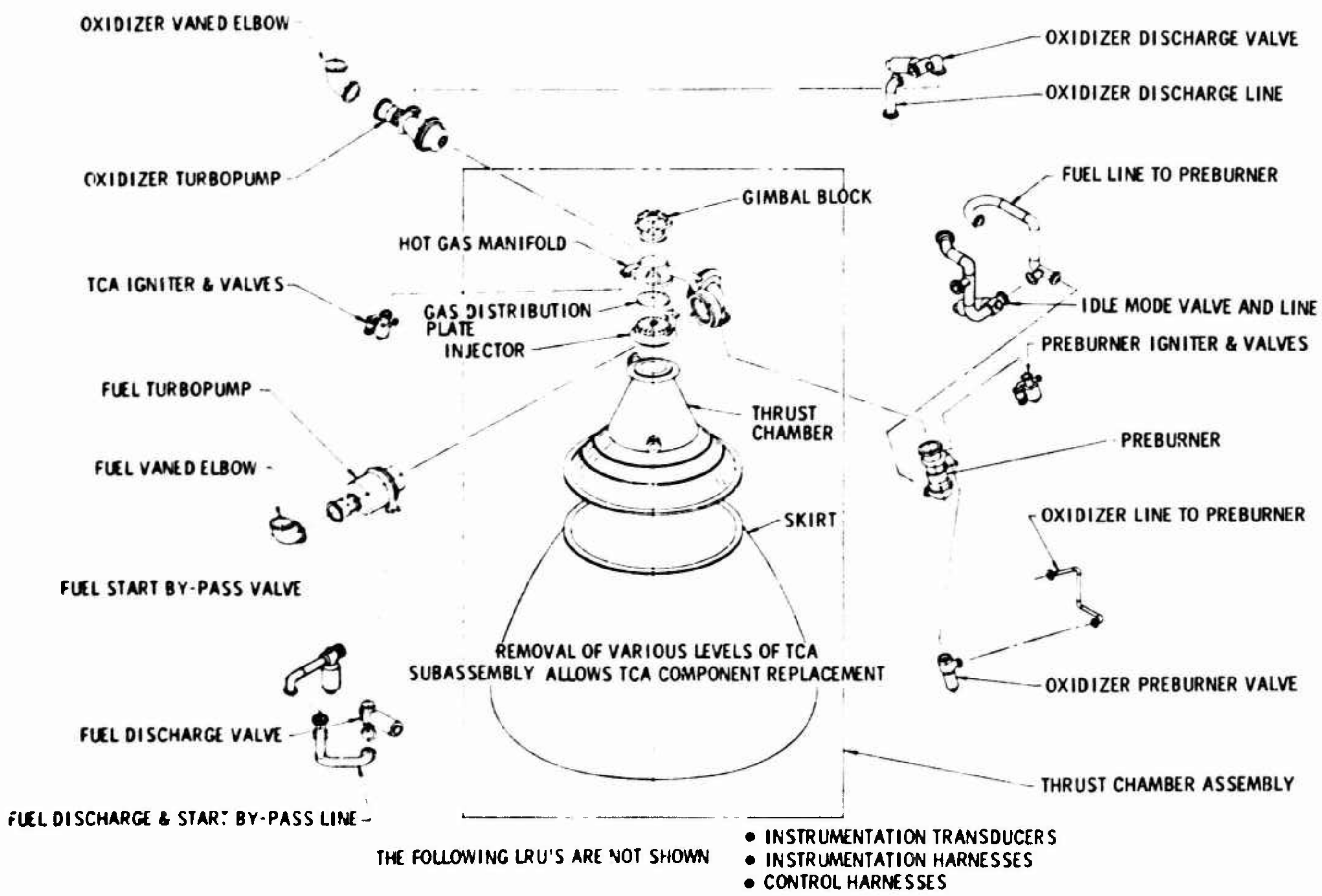

Figure 269. Line Replaceable Units

Through-bolts are employed where possible to reduce problems associated with flange misalignment. Self-locking nuts are used for easier and faster replacement by eliminating the less desirable methods such as safety wire.

Although the turbopump must be removed as a full assembly, turbine components are easily replaced once the TPA has been removed. The TPA is removed from the engine by separation of instrumentation and the propellant inlet and outlet lines and a single turbine hot gas joint. Hot gas inlet and outlet is separated by means of redundant seals which are removed when removing the TPA. Once the TPA has been removed, removal of a single bolt circle allows removal of the turbine nozzles for easy inspection of the turbine rotors. Replacement of the turbine rotor is accomplished by removal of another single bolt circle. Subsequent operations allow easy removal of the last shield and turbine lift-off seal.

The preburner has external bolted flange joints to facilitate easy replacement. Because the preburner walls are regeneratively-cooled the turbine inlet temperature will be measured at the turbine manifold rather than in the preburner. Since the manifold contains a heat shield the instrumentation boss will not "see" the high temperature of the hot gas.

The hot gas manifold is connected to the injector by means of a single bolt circle. The combustion chamber is connected to the injector by means of another bolt circle. The combustion chamber/nozzle is removed as a single unit. The thrust chamber assembly design allows easy removal of the full assembly or any separate part. The major disadvantage of the thrust chamber design is the inability to separate the copper combustion chamber from the tubular nozzle by means of a simple flanged joint. Because of the high heat flux, a brazed (inner-wall) and welded (outer-wall) type of attachment was selected. Even though a simple bolted flange joint was not employed, much consideration was given to this joint due to the low life capability of the combustion chamber and the high life capability of the tubular nozzle. Replacement on the engine level would be accomplished by removal of the thrust chamber assembly (combustion chamber plus nozzle) and a new one installed in its place. After removal and return to the shop area, the combustion chamber would be removed from the nozzle by machining off the flange just upstream of the joint. The outer thrust cone welded joint would be ground out to return it to its original configuration. A heavy shoulder section has been incorporated into the nozzle half to facilitate the grinding operation. The inner wall is 50-mil thick and is cut at an angle so that a 90-mil thick surface is available for mating ease. Once final machining has been accomplished a sheet of braze is installed, the combustion chamber set in place, and the assembly placed in the braze oven for rebrazing. Although the combustion chamber replacement does not allow rapid turn-around reuse of the nozzle, it does give a compromise between reliability versus reusability/maintainability of the nozzle.

2. Space-Based Maintenance

Both complete engine and engine subassembly removal was evaluated for space-based maintenance. For removal of the entire engine the following additional design features must be built into the engine:*

Anchor pads to serve as personal work platforms to allow leverage when removing the suction line bolts and electrical connectors.

Hand-grips on the engine and vehicle to allow emparting a separation force between engine and vehicle and then a retention force (to stop motion). Because of the relative lightweight (500 lbs) of the engine it is felt that two men, one on each side of the engine, could separate the engine and screwjacks would not be necessary.

Double seals with an intermediate collector ring for check-out of the engine after reinstallation. A single fitting from the collector ring would be used in conjunction with a helium leak checker. This method was selected over a pressure decay method because leakage would occur through the pump seals.

Fitting for pressurizing the suction line.

Although not affecting engine design which has capabilities for continuity checks based on ground-based maintenance, a complete engine-to-vehicle electrical check would be required.

Upon comparing complete engine versus engine subassembly replacement for space-based maintenance, it is recommended that complete engine replacement be incorporated. The only exceptions to this are instrumentation transducers which would be replaceable. The complete engine replacement was selected because subassembly replacement would require the following design features:

Anchor pads at many points on the engine system at all LRUs or subassemblys where replacement is desired.

Double seals and collector rings at all LRUs.

Valves must be added or allowances for incorporation of a throat plug after LRU replacement to allow checkout of all components downstream of the pump discharge valves, i.e., preburner, hot gas manifolds, turbines, injector and chamber.

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*Because this study was engine design study, considerations for storing or transporting the engine after removal were not evaluated.

3. Instrumentation Requirements

a. General

Instrumentation requirements are derived for three separate system purposes. These are: (1) flight safety, (2) engine control, and (3) maintenance.

Figure 270 presents a general overlook of the various maintenance concepts and maintenance functions to be considered. The instrumentations required for the OOS Mission are largely dependent on the maintenance concept used. The most desirable OOS Mission is a mission which launches the OOS and recovers the OOS for ground based maintenance after every flight. Depending on the turn-around time of the OOS stage more or less sophisticated instrumentation is required. For short turn-around time, failure mode detection instrumentation and systems analysis are required. This system has the capability of analyzing the engine operational parameters to a standard and flag deviations. For longer turn-around time, this is not required and the operational data would be stored in a data recorder and ground analyzed prior to maintenance allocation. For the case of in-flight maintenance not only must the operational data be analyzed, but also, decision capability has to be provided as to the operational status of the engine which requires sophisticated engine controller computer. Many of the instrumentation parameters can be used for dual or even triple purposes and the parameter list has been selected to allow full use of dual purpose sensors to keep the number of sensors at a minimum. The overall instrumentation list is presented in Table XC and summarized in Table XCI. The system purpose of the instrumentation is also given. The following paragraphs present the reasoning for the instrumentation selection in terms of the function it performs.

b. Flight Safety Instrumentation

Flight safety instrumentation is used to indicate that a safe condition exists. Flight safety monitoring is required both prior to engine start, to assure a readiness condition, and after engine start to prevent engine damage when an out-of-control condition exists. Table XCII presents the flight safety failure mode analysis performed to determine the instrumentation required to assure flight safety. Most of the failure modes can be determined directly, i.e., insufficient pressures, valve not closed, etc. Because reliable ignition detectors do not currently exist, use of valve position and electrical signals are used as an indirect method of determining that ignition has occurred. Not all of the instrumentation used for engine flight safety assurance is supplied by the engine system. Propellant tank pressure which is used to determine adequate Net Positive Suction Head (NPSH) is supplied by the vehicle system.

All flight assurance functions are supplied with separate sets of redundant data. Both sets must indicate an out-of-limit situation before initiation of an engine command will occur. This prevents inadvertant shutdown due to erroneous signals or instrumentation failures.

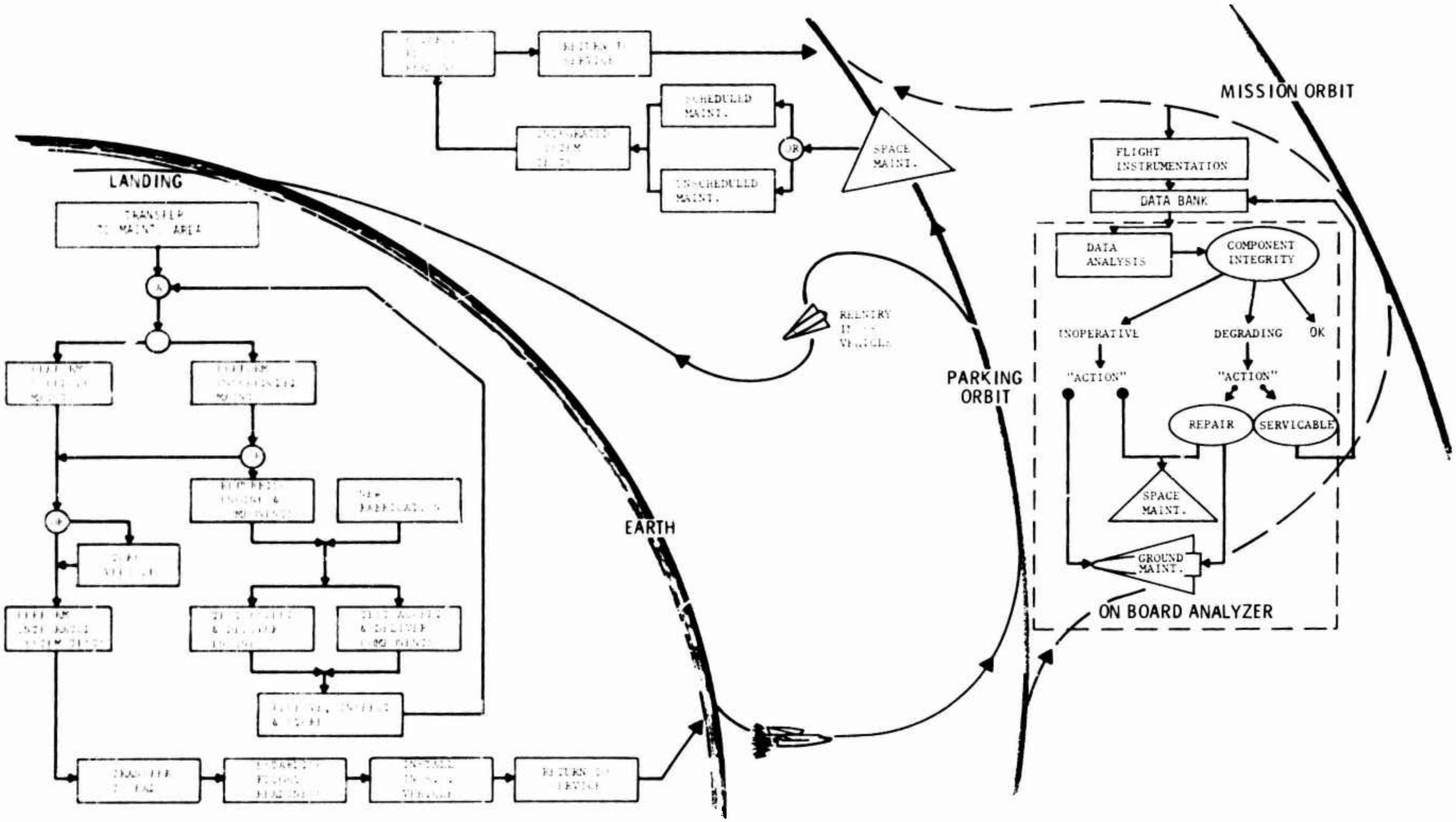


Figure 270. Maintenance Functional Flow

TABLE [illegible]

INSTRUMENTATION LIST

Parameter Number	Parameter	Nomenclature	Purpose F.S.	Purpose C.	Purpose M.	Number of Sensors	Parameter Number	Parameter	Nomenclature	Purpose F.S.	Purpose C.	Purpose M.	Number of Sensors
[illegible]	[illegible]	[illegible]			[illegible]	[illegible]	26	T_{FP}	Fuel Pressurant Temperature			X	1
[illegible]	[illegible]	[illegible]			X	1	27	T_{OP}	Oxidizer Pressurant Temperature			X	1
[illegible]	[illegible]	[illegible]		X		1	28	I_{FPDV}	Fuel Pump Discharge Valve Position			X	1
[illegible]	[illegible]	[illegible] Inlet Pressure			X	1	29	I_{OTCV}	Oxidizer Thrust Chamber Valve Position			X	1
[illegible]	[illegible]	[illegible] Preburner Inlet Pressure			X	1	30	I_{OPBV}	Oxidizer Preburner Valve Position			X	1
[illegible]	[illegible]	Turbine Inlet Pressure			X	1	31	I_{FSBV}	Fuel Start Bypass Valve Position			X	1
[illegible]	[illegible]	Preburner [illegible] Pressure [illegible]		X		1	32	I_{OPBIV}	Oxidizer Preburner Igniter Valve Position			X	1
[illegible]	[illegible]	Turbine Exit Pressure			X	1	33	I_{OTCIV}	Oxidizer Thrust Chamber Igniter Valve Position			X	1
[illegible]	[illegible]	Chamber Pressure	X	[illegible]	X	2	34	I_{PBIM}	Preburner Igniter Monitor Current			X	1
[illegible]	[illegible]	Oxidizer Pump Suction Pressure	X		X	2	35	I_{TCIM}	Thrust Chamber Igniter Monitor Current			X	1
[illegible]	[illegible]	Oxidizer Pump Discharge Pressure - [illegible]			X	1	36	I_{HeTV}	Helium Tank Valve Position	X		X	2
[illegible]	[illegible]	Oxidizer [illegible] Pump [illegible]		X		1	37	A_{FTPA}	Fuel Turbopump Vibration			X	1
[illegible]	[illegible]	Oxidizer Thrust Chamber Inlet Pressure			X	1	38	A_{OTPA}	Oxidizer Turbopump Vibration			X	1
[illegible]	[illegible]	Oxidizer Pump Discharge Pressure - [illegible]			X	1	39	N_{FTP}	Fuel Turbopump Speed	X		X	2
[illegible]	[illegible]	Oxidizer Preburner Inlet Pressure				1	40	N_{OTP}	Oxidizer Turbopump Speed	X		X	2
[illegible]	[illegible]	Fuel Pressurant Pressure			X	1	41	N_{FBP}	Fuel Boost Pump Speed		X		1
[illegible]	[illegible]	Oxidizer Pressurant Pressure			X	1	42	N_{OBP}	Oxidizer Boost Pump Speed		X		1
[illegible]	[illegible]	Fuel Bearing [illegible] Pressure	X			2	43	FMOPB	Preburner Oxidizer Flow		X	X	1
[illegible]	[illegible]	Oxidizer Bearing [illegible] Pressure	X			[illegible]	44	FMOTC	Thrust Chamber Oxidizer Flow		X	X	1
[illegible]	[illegible]	Helium Tank Pressure	X		X	[illegible]							
[illegible]	[illegible]	Helium Purge Cavity Pressure			X	1							
[illegible]	[illegible]	Fuel Pump Suction Temperature	X		X	[illegible]							
[illegible]	[illegible]	Oxidizer Pump Suction Pressure	X		X	[illegible]							
[illegible]	[illegible]	Fuel Preburner Inlet Temperature		X		1							
[illegible]	[illegible]	Turbine Inlet Temperature	X		X	2			Total				56

F.S. = Flight Safety Instrumentation
C. = Control Instrumentation
M. = Monitoring Instrumentation

TABLE XCI

INSTRUMENTATION REQUIREMENTS SUMMARY

Instrumentation System	Purpose	Method
Transient Control and Sequencing	Timing of Sequence	Fixed Time sequence and Feedback Control
	Execution of Sequence	Computer Controlled
Steady State Control	Control to Systems Requirement	Feedback Control System Input
Flight Safety Instrumentation	Flight Safety Assurance	Compare to Safe Operating Limits
Maintenance Instrumentation	Operation Monitoring Data Recording Performance Degradation	Compare to Nominal Performance

TABLE XCII

FLIGHT SAFETY ASSURANCE ANALYSIS

Failure Mode	Failure Effect	Malfunction Indicators
Insufficient Inlet Conditions	Engine Would Not Start	Fuel Suction Pressure Fuel Suction Temperature Oxid. Suction Pressure Fuel Suction Temperature Fuel Tank Pressure Oxid Tank Temperature
Insufficient He	Hazardous Propellant Mixing	Helium Tank Pressure
He Purges In Operation	Hazardous Propellant Mixing	Valve Position
Preburner Ignition	Hazardous Propellant Mixing	Igniter Valve Position Igniter Current
Low Engine Thrust	May Not Accomplish Mission	Chamber Pressure
Fuel Pump Overspeed	Structural Failure	Pump Speed
Oxidizer Pump Overspeed	Structural Failure	Pump Speed
High Turbine Gas Temperature	Structural Failure	Preburner Chamber Temperature
Loss of Bearing Coolant Flow	Bearing Failure	Down Stream Coolant Pressure
Premature Main Fuel Valve Closure	Serious Engine Damage	None*

*ΔP Interlock to prevent premature closure

c. Control Instrumentation

When utilizing a closed-loop control system, both mixture ratio and thrust must be continuously monitored. Since neither of these are direct measurements, these are calculated within the engine controller from flowmeter, temperature, and pressure measurements. Table XCIII presents a listing of parameters used to obtain mixture ratio and thrust.

From Table XCIII it is seen that redundant mixture ratio "measurements" are obtained by two separate and different methods. The primary method is to use oxidizer flowmeters for the oxygen side and the pressure drop across the preburner injector for the hydrogen side. Although normally both pressure and temperature are measured upstream of the flowmeter to determine density, it is assumed the oxygen temperature can be estimated with sufficient accuracy based on the pressure. On the fuel side, this is not true, since the hydrogen will leave the chamber cooling jacket and enter the preburner in the gaseous state.

The backup method for determining flow rates and mixture ratio is obtained from boost pump head rise measurements and speed and a knowledge of the head-capacity relationship. A backup method different than the primary method for determining mixture ratio was employed due to the limited space available for flowmeters. Use of the boost pumps was not selected as the primary method because it eliminates the maintenance failure prediction capability for the boost pump, i.e., since flow is calculated from head and speed, it would always indicate the correct head-flow-speed relationship.

Thrust is obtained by measuring chamber pressure and then calculated based on throat diameter, area ratio, mixture ratio, and flow. Redundancy is obtained by use of two chamber pressure measurements.

d. Maintenance Instrumentation

Maintenance instrumentation is used to isolate a Line Replaceable Unit (LRU) failure and/or predict when a failure is about to occur. The instrumentation requirement is based upon the failure modes and effects analysis (see Table LXXXII of Section III.B.6). Since the maintenance measurements are not used to generate command signals to the engine, redundancy was not incorporated and therefore a single measurement is employed for each function. Prior to LRU replacement, instrumentation accuracy will be obtained either by direct test and/or evaluation of other engine system parameters.

TABLE XCIII

CONTROL SYSTEM INSTRUMENTATION

Variable Controlled	Method of Control	Parameter Measured
Mixture Ratio	Measure engine flow, convert to mixture ratio, and compare to desired level.	**Primary** Preburner Oxidizer Flowmeter Thrust Chamber Oxidizer Flowmeter Oxidizer Pump Discharge Pressure - T.C. Oxidizer Pump Discharge Pressure - P.B. Preburner Fuel Injector Pressure Drop Fuel Preburner Inlet Pressure Fuel Preburner Inlet Temperature **Backup** Oxidizer Boost Pump Head Rise Oxidizer Boost Pump Speed Fuel Boost Pump Head Rise Fuel Boost Pump Speed
Thrust	Measure Chamber Pressure Calculate Thrust from Chamber Pressure and Mixture Ratio above.	Chamber Pressure

D. ENGINE DEVELOPMENT PLANS AND COST (25K ENGINE DESIGN)

1. Program

The approach to OOS program planning differs from past industry practice. Historically, plans have been established in response to rigid contractual requirement -- typically, with compressed schedules dictated by pressing national objectives.

The plans which form the basis for the cost studies attempt to recognize the current fiscal climate and the attendant emphasis on economy rather than crash programs.

The program span has been somewhat arbitrarily established as ten years. The Demonstrator Engine portion of the program is planned for the first five years.

The Demonstrator Engine Program will start with preliminary design activities to define critical technology areas. These areas will then be investigated in a comprehensive series of laboratory type tests. The next phase consists of the fabrication and test of one demonstrator engine. The primary purpose of this engine will be to demonstrate the adequacy of the selected design concept. Since no unrealistic stringent design goals must be satisfied, (e.g., highest possible performance or minimum weight) only limited hardware will be required, which is in consonance with the overall fiscal policy.

The final portion of the Demonstrator Engine Program involves transforming the knowledge gained in the two previous phases into working drawings and documents which would form the foundation for the development program.

The development portion of the program consists of three phases: Block I, Preliminary Flight Certification (PFC); and Final Flight Certification (FFC). Block I efforts will be accomplished within three program years, with one year allocated to the PFC and FFC phases, respectively.

The Block I effort will be a logical extension of the Demonstrator Engine Program. Component development based upon the results of the Demonstrator Engine Program will be accomplished during the first two years of the development program. It will be noted that only two design iterations are planned for this period, with relatively limited amounts of hardware. This approach is considered feasible because of the benefits accrued from the Demonstrator Engine Program which permitted a methodical approach to the solution of problems at the subcomponent level. Consequently, the problem solving effort during component development should be materially reduced from the level experienced on more accelerated programs.

A soft mockup will be fabricated early in Block I. This will permit the precise definition of interfaces, clearances, and line routings. It precludes the need for most of the time consuming layouts that would

be required to define interface locations in space and minimizes costly human error. The soft mockup will also eliminate most of the costly, time consuming iterations normally associated with first article assembly, i.e., connection with ancillary lines, harness routing, clearance of lines and harnesses handling lug location, and bracketry installation.

The Block I portion of the Development Program is concluded during the third program year with engine level tests both at altitude and at sea level. The objective of these tests will be to demonstrate attainment of performance goals. It is anticipated that "tune up" type modifications must be implemented to achieve these goals. Six engines are planned to support the test activity -- three at each test site. One engine will be in the test stand, one in backup, and one undergoing refurbishment at any given time.

Immediately after completion of the Block I test program, a design freeze will be implemented preparatory to entering the flight certification phase of the program.

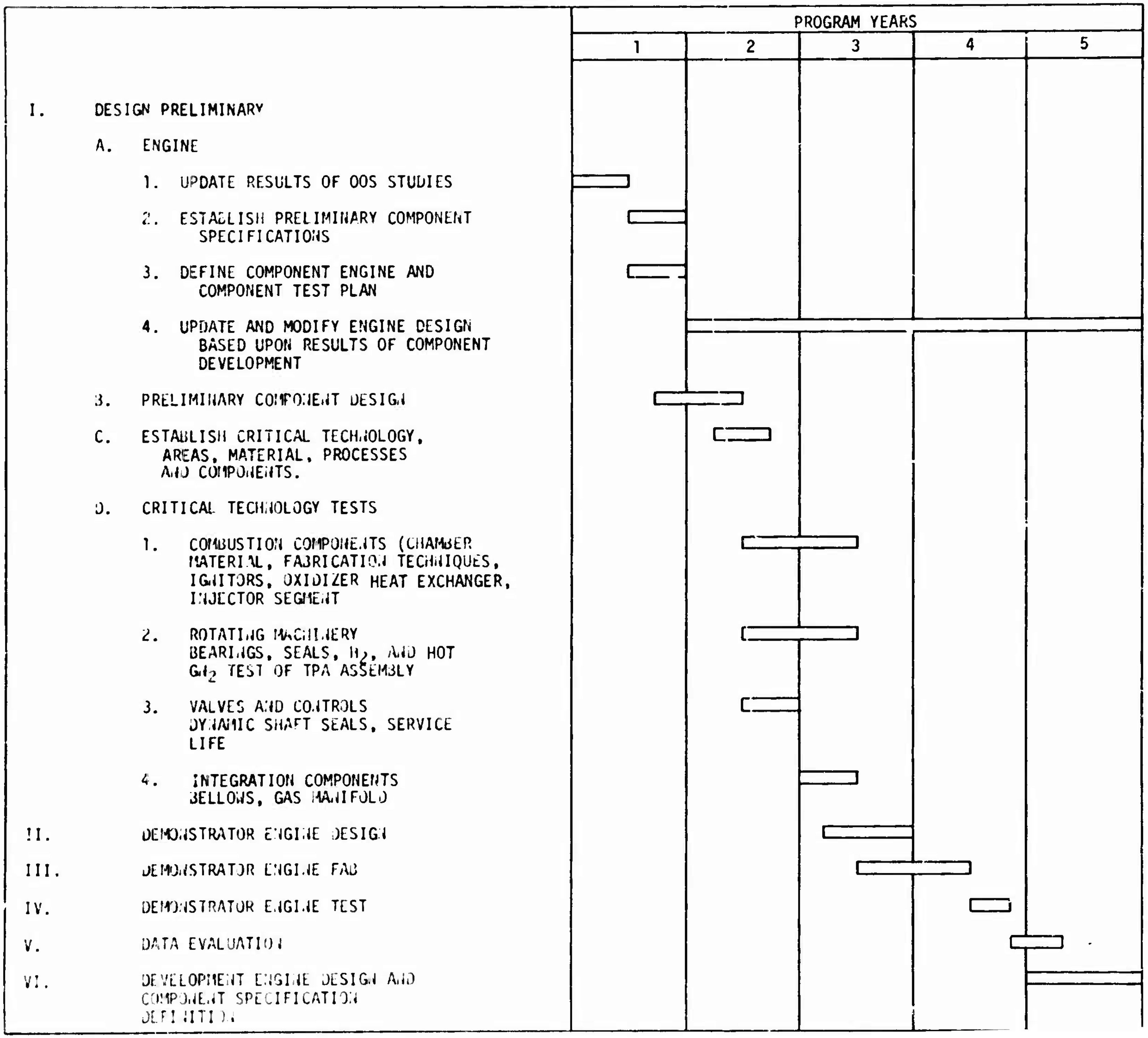
Production type drawings, specifications, tooling and controls will be used during the fabrication and testing of four each PFC and FFC engine assemblies to insure that the test articles are identical to the production engines.

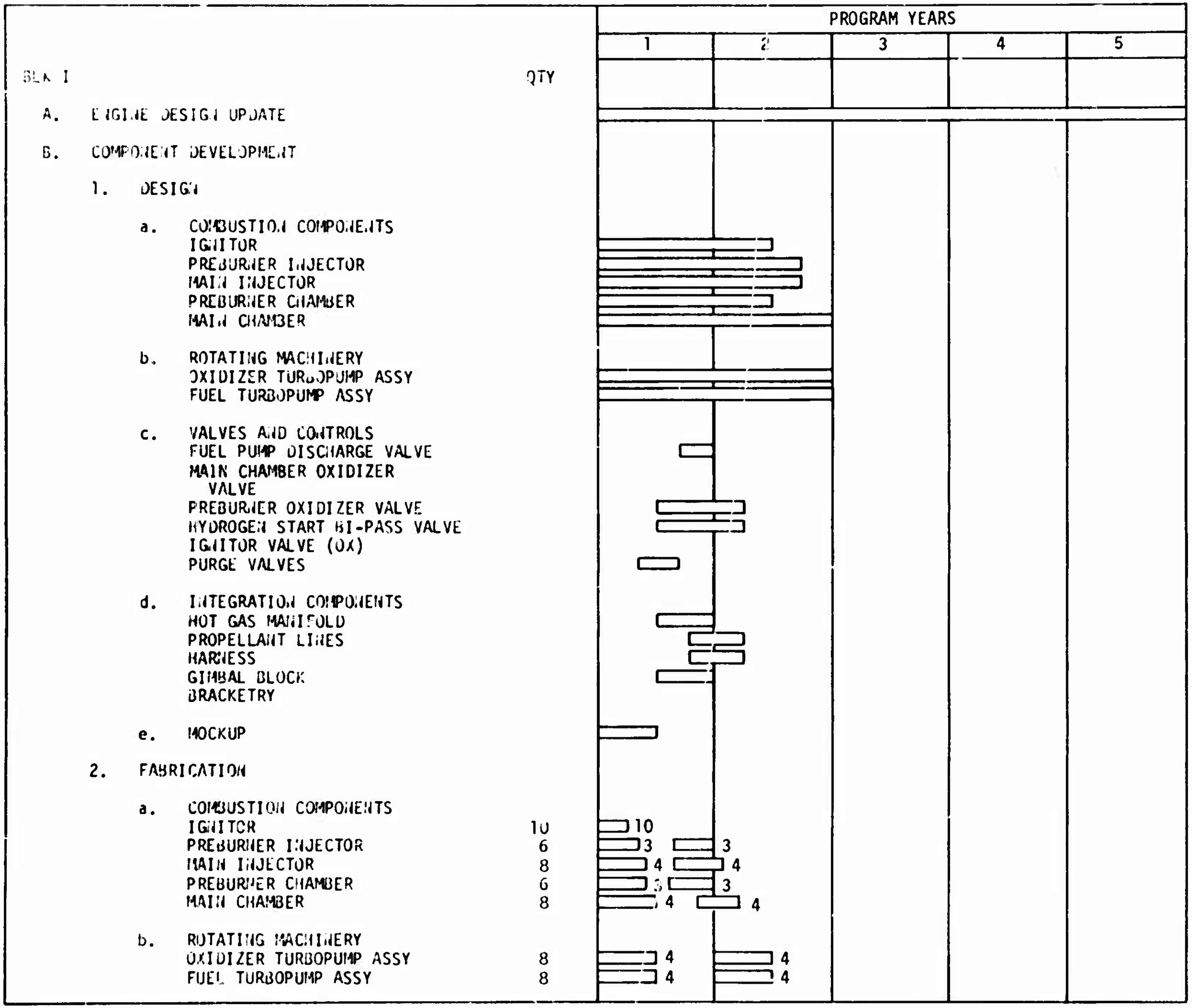
Preliminary flight certification will be conducted primarily to demonstrate the safety aspects of the engine with normal performance testing. Additionally, the previous Block I history will be reviewed to ensure that identified failure mechanisms have been corrected.

Final flight certification tests represent the satisfactory completion of the development program as well as formal demonstration that all of the design requirements collected from every facet of the development program have been met.

The final program activity is the initiation of long lead procurement for the production engine. This activity starts concurrent with the FFC tests. This start time was selected because; (1) the adequacy of the engine has already been well demonstrated, and as a consequence, little risk of premature ordering exists, and (2) experienced personnel are available to staff the program office.

The schedules for both the Demonstrator and Development portions of the program are included in Figures 271 and 272.


PROGRAM YEARS
1
2
3
4
5
I. DESIGN PRELIMINARY
A. ENGINE
1. UPDATE RESULTS OF OOS STUDIES
2. ESTABLISH PRELIMINARY COMPONENT SPECIFICATIONS
3. DEFINE COMPONENT ENGINE AND COMPONENT TEST PLAN
4. UPDATE AND MODIFY ENGINE DESIGN BASED UPON RESULTS OF COMPONENT DEVELOPMENT
B. PRELIMINARY COMPONENT DESIGN
C. ESTABLISH CRITICAL TECHNOLOGY, AREAS, MATERIAL, PROCESSES AND COMPONENTS.
D. CRITICAL TECHNOLOGY TESTS
1. COMBUSTION COMPONENTS (CHAMBER MATERIAL, FABRICATION TECHNIQUES, IGNITORS, OXIDIZER HEAT EXCHANGER, INJECTOR SEGMENT
2. ROTATING MACHINERY BEARINGS, SEALS, H2, AND HOT GH2 TEST OF TPA ASSEMBLY
3. VALVES AND CONTROLS DYNAMIC SHAFT SEALS, SERVICE LIFE
4. INTEGRATION COMPONENTS BELLOWS, GAS MANIFOLD
II. DEMONSTRATOR ENGINE DESIGN
III. DEMONSTRATOR ENGINE FAB
IV. DEMONSTRATOR ENGINE TEST
V. DATA EVALUATION
VI. DEVELOPMENT ENGINE DESIGN AND COMPONENT SPECIFICATION DEFINITION


PROGRAM YEARS
1
2
3
4
5
BLK I
QTY
A. ENGINE DESIGN UPDATE
B. COMPONENT DEVELOPMENT
1. DESIGN
a. COMBUSTION COMPONENTS
IGNITOR
PREBURNER INJECTOR
MAIN INJECTOR
PREBURNER CHAMBER
MAIN CHAMBER
b. ROTATING MACHINERY
OXIDIZER TURBOPUMP ASSY
FUEL TURBOPUMP ASSY
c. VALVES AND CONTROLS
FUEL PUMP DISCHARGE VALVE
MAIN CHAMBER OXIDIZER VALVE
PREBURNER OXIDIZER VALVE
HYDROGEN START BI-PASS VALVE
IGNITOR VALVE (OX)
PURGE VALVES
d. INTEGRATION COMPONENTS
HOT GAS MANIFOLD
PROPELLANT LINES
HARNESS
GIMBAL BLOCK
BRACKETRY
e. MOCKUP
2. FABRICATION
a. COMBUSTION COMPONENTS
IGNITOR 10
PREBURNER INJECTOR 6
MAIN INJECTOR 8
PREBURNER CHAMBER 6
MAIN CHAMBER 8
b. ROTATING MACHINERY
OXIDIZER TURBOPUMP ASSY 8
FUEL TURBOPUMP ASSY 8
10
3
3
4
4
3
3
4
4
4
4
4
4

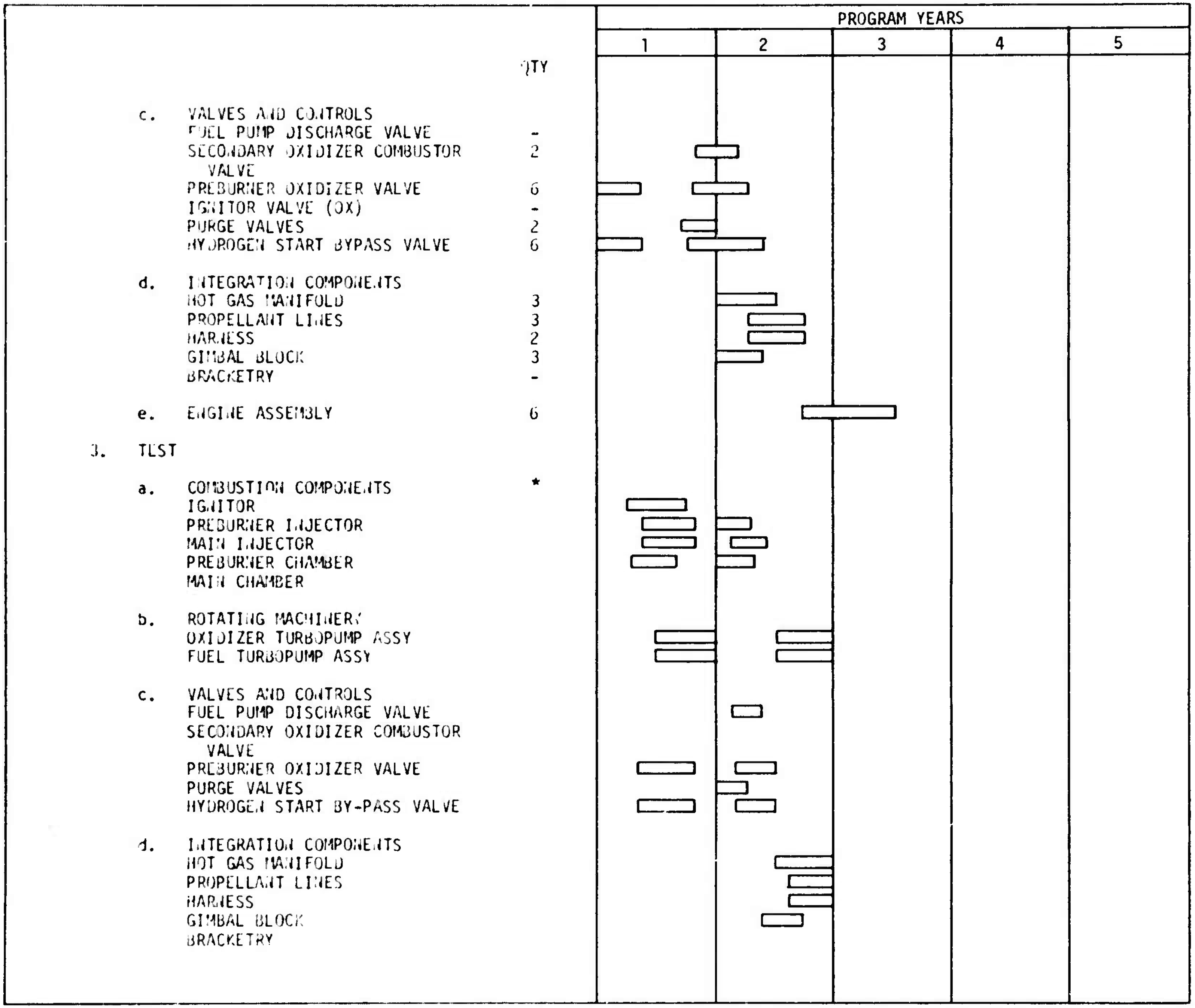


Figure 272. Development Engine Program Schedule (2 of 3)

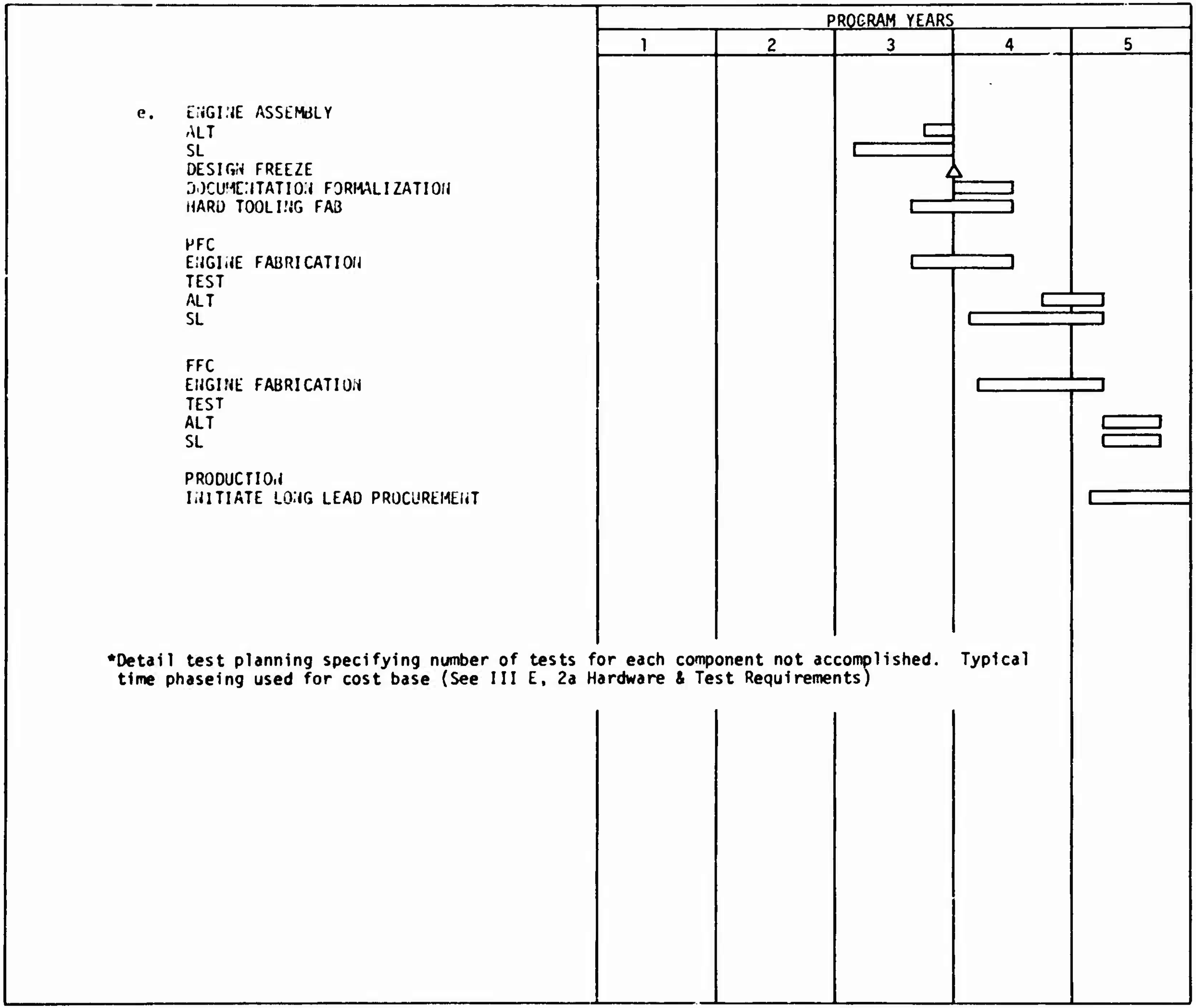


Figure 272. Development Engine Program Schedule (3 of 3)

2. Hardware and Test Requirements

a. Hardware

The limited contract funding available for program planning and cost estimation (<5%) made it impractical to attempt detailed hardware demand and test schedules. In lieu of detailed plans, the approach has been to relate to similar programs.

Estimating the hardware requirements of the Demonstrator Engine Program is difficult because much of the hardware will be raw material and sub-components to support laboratory type tests. Many of the tests will be spin-offs from previous tests. Consequently, hardware requirements have been simplified to two equivalent engines. This level is similar to that experienced on similar programs.

Hardware requirements for the Development Engine Program are superimposed upon the fabrication portion of the program schedule (Figure 272). The quantities as shown in the component development portion are sufficient to support two complete design iterations. Six Block I engines will be fabricated and tested before design freeze. After the design freeze, four engines each are scheduled for the PFC and FFC phases.

The prime reference source used to determine the foregoing hardware requirements was the planning accomplished under the auspices of Contract NAS8-26188 for the AJ-550 Space Shuttle Main Engine.

When comparing the requirements of that program with those of the OOS, it will be noted that component development requirements of the OOS are less and engine requirements are almost identical.

It is assumed less effort will be required for component development because of the knowledge gained from the Demonstrator Engine Program of the OOS Program. Approximately the same number of engines are required because both engines must be capable of being man-rated and as a consequence, must have similar test histories.

b. Test

(1) Program

The criteria for the number of tests to be conducted on the engines and in each component area will be based upon the accomplishment of specific objectives. Factors which influence total number of tests in any component test series include complexity of objectives and capability to plan accomplishment of multiple objectives of a given test. As has already been mentioned, funding and time limitations precluded extremely detailed planning, however, the general philosophy and type of test envisioned for the major engine components are known and are summarized in the following paragraphs.

Demonstrator engine testing is initiated with components designed to satisfy analytically defined system requirements. Prior to initial engine test, materials and components are tested both individually and as part of assembled subsystems to verify their adequacy to proceed. Prime importance during these tests is placed upon gaining confidence that the design can be committed to initial engine testing. Therefore, the first series of engine tests will be structured to identify nominal engine system environmental conditions and functional interactions to demonstrate the general adequacy of the engine cycle rather than maximum performance. The evaluation of these data and the resultant engine design will constitute the basis for the Block I engine of the Development Engine Program. The test level of effort assumed for costing was two years for the physics and engineering laboratories followed by two years in the test area.

The Engine Development Program is based upon the redefined design and test requirements for the components from the analysis of the Demonstrator engine test data. Testing at both sea-level and altitude will include environmental conditioning, fail-safe aspects, and stability evaluation of system capabilities. Subsequent exposure through the PFC program will permit an assured progression through a valid Final Flight Certification Program. The test level of effort assumed for the Development Phase was 1 year laboratory level activity, 3 year ALRC test area activity, and 1.25 years at AEDC.

(2) Component

(a) Turbomachinery

Prerequisites to turbomachinery development testing are:

- Detailed design, structural, material, performance, reliability, maintainability, and producibility analyses.

- Comprehensive master layouts detailing tolerance variations and effects.

- Use of "rig testing" for evaluation of components to define those physical phenomena not subject to analysis and to validate component capability.

The higher assembly testing commences at the earliest possible date as determined from component rig test demonstrations of acceptable attributes. Rig and assembly testing are not predicated on complete success. Iterations and contingencies are expected and planned for at all levels of testing. The assembly testing then affords validation or

redefinition of the component requirements. Requirements validation will permit continued assembly testing to more stringent levels of operation. Redefinition will require additional component evaluation if the requirements exceed the determined or design capabilities.

(b) Combustion Devices

The initial and all subsequent iterations of the combustion components will be subjected to a three-point development approach. Static testing will be primarily oriented toward evaluation of all-inclusive worst-case conditions. Laboratory testing will be used to the fullest extent to yield low cost non-firing data, particularly during the Demonstrator Engine Program. Structural tests will expose design weak points, define structural failure modes and, through test-to-failure, assess design margins. Satisfactory development of the preburners to the prescribed level will constrain their use for Thrust Chamber Subsystem (TCSS) testing. TCSS testing permits progressive development of the main injector, chamber, nozzles, and hot gas manifold.

(c) Valves

The valve program is sequentially oriented like all of the other contributing efforts. However, there is an early need for particular units; therefore, a priority is established for the development of the preburner control valves (oxidizer and fuel), the igniter oxidizer valves, the main oxidizer and fuel by-pass valves, and the electromechanical actuators. These components are constraints to component development testing, which are, in turn, constraints to engine system testing. The remaining controls components are required for engine testing at a later date.

The valve and actuator design and development activity is categorized into the following distinct phases, at the end of which the valves will have demonstrated capability for performing in the operational flight program:

Preliminary Testing: This consists of testing during the Demonstrator Engine Program of commercially-procured or fabricated individual components and subcomponents for design requirements validation and to establish design capabilities. Extensive use will be made of overstress testing techniques, thermal exposure, endurance cycling, and life-proof loading.

Development Testing: The primary objective is to determine the adequacy for engine and subsystem testing of those units evolved from the Preliminary Testing. The design analysis techniques developed during the preliminary

testing will be utilized to the fullest extent in defining this design level. Primary categories of testing will be cyclic response, flow, pressure, endurance, and induced as well as natural environments.

Preliminary Verification and Peripheral Testing: These tests will be conducted to evaluate the revised designs and test requirements emanating from the engine system test data. The objectives are to validate the second generation design adequacy for continued engine system usage and entry into component verification. These tests are planned for the second iteration of the Block I component test series.

PFC: In addition to support of the engine and other component testing, valves will be tested in malfunction modes deemed inadvisable to perform on the engine, as part of the flight safety evaluation.

FFC: The final testing program primarily is associated with the endurance and cycle life capabilities of the valve designs.

(d) Harness and Instrumentation

The electronic engine controller development is assumed to be subcontracted. The major system development effort will be conducted within ALRC laboratories and in conjunction with other component, subsystem, and engine system scheduled development testing. Software development will be concurrent with these activities.

Design requirements definition will be accomplished by using a breadboard version of the controller. This activity will provide precise circuit design requirements for functional performance of closed-loop control, stored start and stop sequences, flight safety, fault isolation, and engine systems maintenance data.

The harnesses and instrumentation will be subjected to extensive laboratory evaluation, plus use during all applicable component and engine system testing.

(e) Engine Integration Components

The components are the gimbal assembly and engine interconnect lines system. The majority of the design evolution will occur in conjunction with the engine testing.

The early portion of the program for these components is devoted to testing of subcomponents or specimens to define the analytical aspects and characteristics under imposed test conditions. This definition then is applied to the design of units which undergo testing for validation of criteria and demonstration of adequacy for engine use. Definition of engine operating environment, design criteria revision, and retesting to the revised requirements provides component upgrading to the level required for component verification and engine system certification.

3. Facilities and GSE

No new facilities are required to fabricate or test the OOS Engine. During the early portion of the Demonstrator Engine Program, much of the work will be accomplished at the laboratory level. ALRC has well equipped nondestructive test laboratories to support all of the work now contemplated. Later in the program, the ALRC Aerophysics Laboratory will be used in tests of small combustion devices. The Demonstrator Engine components will be fabricated in the Research and Development Manufacturing complex and engine testing will be accomplished in the ALRC J-area test stand.

Fabrication of engine components will be shifted to the regular ALRC fabrication facilities during the Development Program. Engine tests are planned both for ALRC J-area and Arnold Engineering Development Center, Tullahoma, Tenn.

Modifications to the ALRC test facility are listed in Table XCIV and GSE requirements are tabulated in Table XCV.

4. Propellant Requirements

The propellant requirements were estimated in accordance with the established ALRC practice of projecting quantities on the basis of time in the test area for an engine of a given thrust level. Experience has demonstrated that because of the various usage factors (boil off, spillage, contamination etc.) this approach is more satisfactory than ordering for a specific run duration.

Propellant requirements for the Demonstrator and Development Engine Programs are:

	Demonstrator	Development
LH_2	1,110,000 lb	5,910,000 lb
LO_2	3,320,000 lb	17,646,000 lb
LN_2	663 ton	3,532 ton
H_e	1,200 KSCF	6,400 KSCF

5. Project Control Methods

The project control methods shown in Figure 270 with reference to the program logic diagram are self explanatory.

It will be noted that controls during the Demonstrator Engine portion of the program are of the informal variety in order to provide maximum engineering flexibility.

TABLE XCIV

FACILITIES REQUIREMENTS

Combustion Component Test

Propellant Run Lines 4" (Install)

GH_2 Vent Stacks (Relocate Existing)

Thrust Fixture Fabricate

GN_2 Cascade 4500 psi (Relocate)

GH_2 Cascade 4500 psi (Relocate)

GN_2 and GH_2 Converters (Relocate)

Instr. and Controls Systems (Install)

TPA Test

Low Pressure Tanks (Relocate Existing)

Propellant Lines (Relocate Existing)

Fixture (Relocate Existing)

Instr and Control (Relocate Existing)

Engine Test

Diffuser System Modification

Chamber Mode Modification

Fixture Modification

Prop Run Piping Modification

LH_2 Run Vessel (Use Storage) Modification

Instr and Control Modification

TABLE XCV

CHARACTERISTIC GSE REQUIREMENTS

	Units
Transport and Handling	
Shipping Containers	40
Handling Frame	2
Installation and Removal Set	3
Sling Set Component	3
Sling Set Nozzle	2
Stiff Links Gimbal	4
Sling Engine Handling	2
Trailer	4
Safety and Protective	
Kit Engine Protective Covers	44
Cover Environmental	10
Kit TPA Protective Closure	2
Kit Hot Gas Manifold	2
Kit Valve Protective	5
Inspection, Test	
Kit Engine Leak Detector	2
Kit Engine Leak Test Closure	2
Simulator Engine Sensor	2
Simulator Engine Valve	2
Inspection Unit Fiberoptic	2
Seal Assembly, Comb Cham Thrust	20
Inspection Set Main Inject Face	2
Manifold Seal	2
Test Set Staff Displ	4

TABLE XCV (cont.)

Shop Maintenance and Service	Units
Stand Maintenance LPTPA	2
Tool LPTPA	2
Tool Kit Valve	2
Stand Engine	2
Stand Maintenance HOTPA	2
Stand Maintenance HFTPA	2
Tool HFTPA and HOTPA	4
Lapping Kit	1
Test Set Cryo/Pneumatic	1
Test Set Igniter	1

The controls gradually become more stringent in the Development Engine portion until all of the various disciplines are imposed during the fabrication of the PFC engines. These controls constitute one of the inputs used to project manpower load which was in turn used as one of the elements of cost.

6. Methods of Cost Estimating

The basic contract required that program planning and cost estimating be accomplished with no more than 5% of the total program effort. Consequently, it was impractical to attempt a detailed cost analysis of each of the engine 47 variations considered.

The approach to costing was as follows:

1. A representative schedule and program plan was established as described in Section III.E.1.

2. Hardware requirements were estimated based upon experience with similar programs. (Quantities are shown on the program schedule, Figure 272).

3. Test and propellant costs were estimated based upon typical test area facility requirements and the program schedule.

4. Hardware fabrication costs were estimated based upon available drawings (Table XCVI). Total hardware costs were then developed from the hardware demand portion of the program schedule (Figure 272).

5. Engineering manpower costs were based upon the engineering manpower distribution spread shown in Figure 273, which was generated pursuant to the requirements of the program controls shown in Figure 274.

The foregoing cost elements formed the basis for the costs of the 25K baseline engine shown in Figure 275.

Costs for design and thrust variations were then estimated as a delta to the base costs. The mechanics of this process required estimates from the Engineering, Manufacturing and Test areas relative to the per cent change in complexity for each of the various conditions, including configuration variations shown in Figures 275 and 276. The algebraic sum of the inputs defined the cost variation.

The results of the study are summarized in Tables XCVII through XCIX.

As shown, all costs fall within a relatively narrow band. This is because engineering and support manpower costs are the major cost elements in any program and specific design details exert a relatively minor cost impact upon the total program.

TABLE XCVI

COMPONENT COST BREAKDOWN

Component	Demonstrator	Development	Production
Thrust Chamber Nozzle (Regen)	$ 112,621	$ 71,429	$ 65,656
Injector	53,000	30,727	26,904
Hot Gas Manifold	66,245	36,297	31,937
Gimbal Block	13,572	4,283	3,834
Oxidizer TPA (Incl Boost Pump)	48,236	42,628	41,826
Fuel TPA (Incl Boost Pump)	76,774	70,550	69,589
T.C. Igniter and Valve	17,772	11,242	10,376
P.B. Igniter	7,000	4,428	4,087
Preburner	30,000	17,126	14,691
Oxidizer Preburner Valve	29,732	11,595	10,281
Oxidizer Discharge Valve	33,115	16,519	14,498
Fuel Start Bypass Valve	33,115	16,519	14,498
Fuel Discharge Valve	28,000	14,491	12,816
Oxidizer Vaned Elbow	750	750	750
Fuel Vaned Elbow	750	750	750
Turbine Bypass Valve	33,115	16,519	14,498
Fuel Discharge and Start Bypass Line			
Fuel Discharge Line			
Fuel Line to Preburner	33,019	16,985	15,125
Oxidizer Line to Preburner			
Power Head Assembly	11,863	7,921	6,477
Thrust Chamber Assembly	25,873	16,036	12,443
Electrical Harnesses	11,200	11,200	11,200
Sensors	39,900	39,900	39,900
Controller	125,000	125,000	125,000

PROGRAM YEAR	1	2	3	4	5	6	7	8	9	10
PROGRAM OFFICE										
PROGRAM MANAGER	1/1 OVERHEAD									
CONTRACT ADMINISTRATION	1/2/0 OVERHEAD					3/1				
FISCAL CONTROL	1/2/0			1/1/2		8/1				
RELIABILITY	1/2/0					3/1		6/1		
CONFIGURATION MANAGEMENT						3/1	6/1			
DOCUMENTATION	1/2/0				1/0	1/4	4/4			2/4
ENGINEERING										
ENGINEERING MANAGER	1/1					2/1				
ENGINE ASSY &	1/1/2		3/3			6/4				
INTEGRATION SUPPORT						2/			1/0	1/2
PERFORMANCE & ANALYSIS	1/			2/		4/				
COMBUSTION COMPONENTS	3/3		5/8	5/3		10/8		8/2	6/1	
SUPPORT	6/1		3/1			6/1	2/1			
ROTATING MACHINERY	4/3		6/8	4/3		10/8		8/2	6/1	
SUPPORT	2/1		3/1			6/1	2/1			
VALVES & CONTROLS	3/1		7/4	3/1		10/8		8/2	6/1	
SUPPORT	1/2/2		3/0		1/0	2/1	1/2/1/2			

PRIME FUNCTION ENGR/DRAFTING
SUPPORT FUNCTION STRESS, THERMO, ETC./LABORATORY TECHNICIANS

Figure 273. Engineering Man Loading

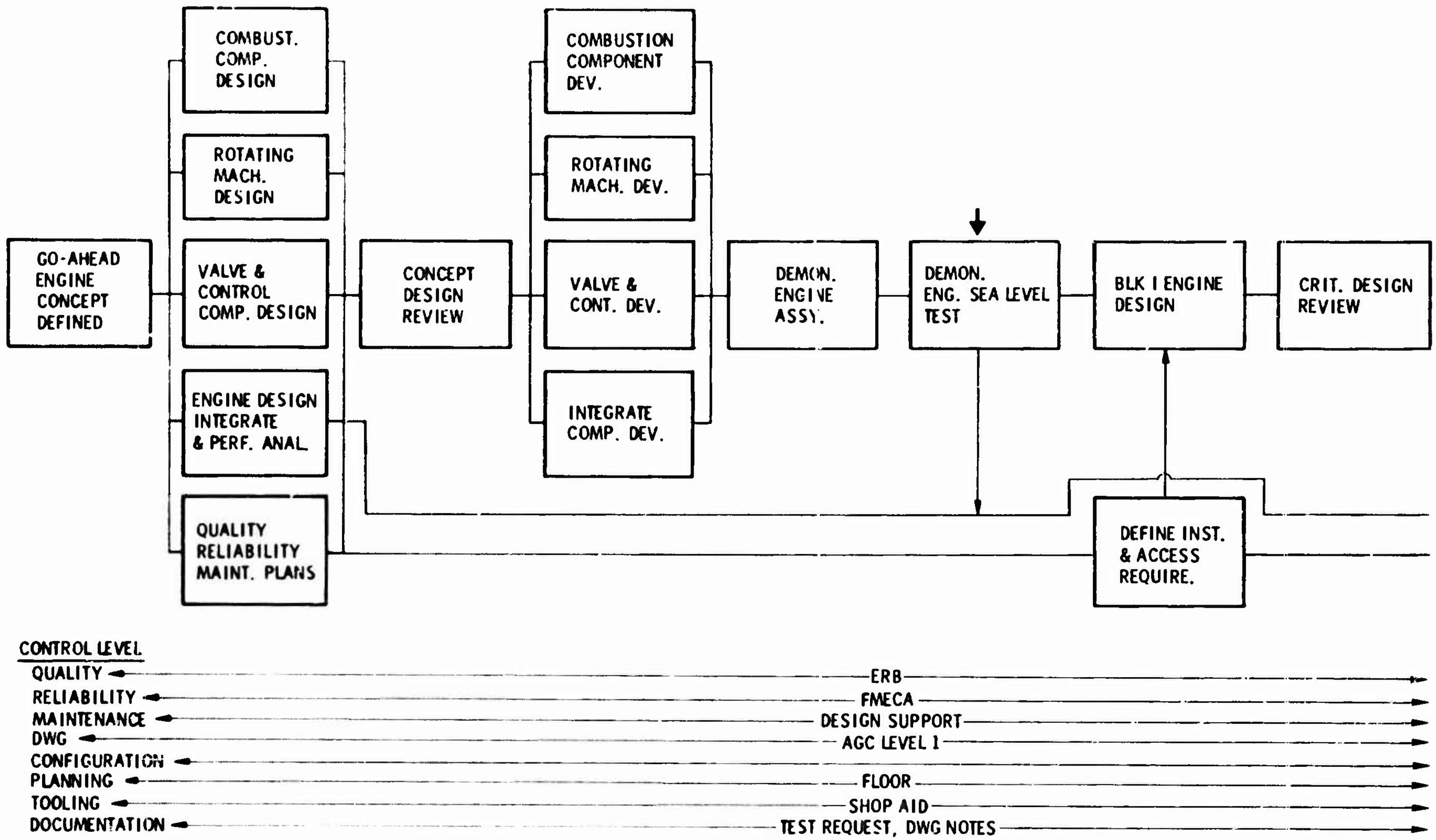
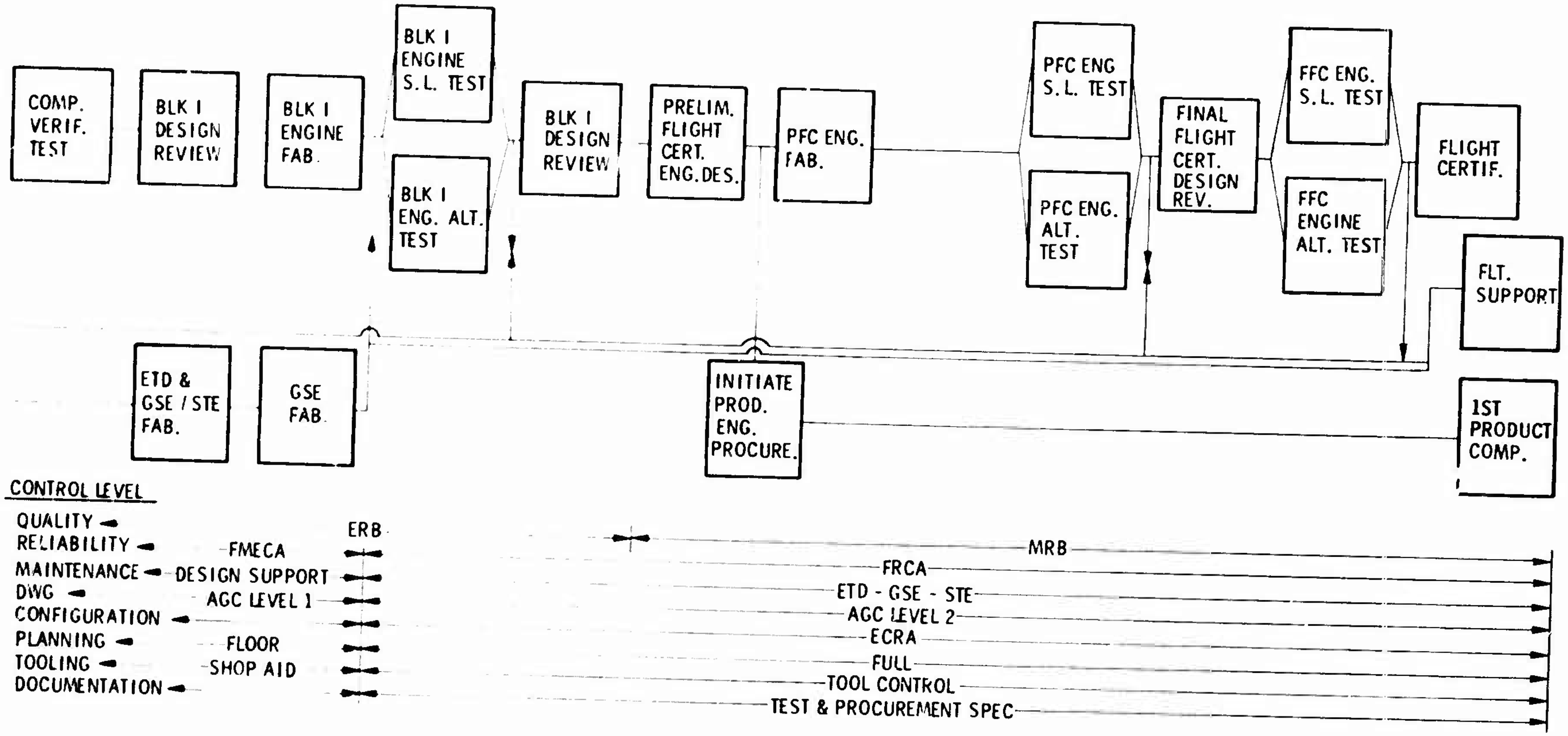


Figure 274. Program Control Methods (Sheet 1 of 2)



DEFINITIONS

ERB	ENGINEERING REVIEW BOARD
MRB	MATERIAL REVIEW BOARD
FMECA	FAILURE MODE AND EFFECT CRITICALITY ANALYSIS
FRCA	FAILURE REPORTING AND CORRECTIVE ANALYSIS
ETD	ENGINEERING TEST DIRECTIVE
GSE	GROUND SERVICE EQUIPMENT
STE	SPECIAL TEST EQUIPMENT
ECRA	ENGINEERING CHANGE RELEASE ACTIVITY

Figure 274. Program Control Methods (Sheet 2 of 2)

SUBSYSTEM		GAS GENERATOR CYCLE				TCA TAP-OFF				COOLANT TAP-OFF			
	F	8K	15K	25K	50K	8K	15K	25K	50K	8K	15K	25K	50K
	P_c	1100	1150	1250	1400	1100	1150	1250	1400	750	900	1000	—
GIMBAL SYSTEM :													
BOOST PUMP: LO_2		NO	NO	NO	NO	NO	NO	NO	NO	NO	NO	NO	NO
LH_2		NO	NO	NO	NO	NO	NO	NO	NO	NO	NO	NO	
TPA: LH_2 PUMP		2	2	2	2	2	2	2	2	1	2	2	
LH_2 TURBINE		2	2	2	2	2	2	2	2	2	2	2	
LO_2 PUMP		1	1	1	1	1	1	1	1	1	1	1	
LO_2 TURBINE		2	2	2	2	2	2	2	2	2	2	2	
PREBURNER OR GAS GENERATOR		YES	YES	YES	YES	NO	NO	NO	NO	NO	NO	NO	NO
IGNITION SYSTEM		2	2	2	2	1	1	1	1	1	1	1	1
ELECTRIC SYSTEM		SAME →											
INSTRUMENTATION		SAME →											
PURGE SYSTEM		SAME →											
CHILLDOWN SYSTEM		TCA BP →											
NOZZLE RETRACTION		NO	NO	YES	YES	NO	NO	YES	YES	NO	YES	YES	YES
ENGINE CONTROL		LV	LV	LV	LV	HGV	HGV	HGV	HGV	CGV	CGV	CGV	CGV
ENGINE WEIGHT	F/P_c	7.3	13.0	20.0	35.7	7.3	13.0	20.0	35.7	10.7	16.66	25.0	—
	ϵ	450	380	450	320	450	380	450	320	340	450	440	
	Wt	200	300	520	860	200	300	520	860	230	410	580	

Figure 275. Configuration Definition

SUBSYSTEM			STAGED COMBUSTION				STAGED COMBUSTION-BLEED				EXPANDER			
		F	8K	15K	25K	50K	8K	15K	25K	50K	8K	15K	25K	50K
		P_c	1300	1450	1800	2200	1300	1450	1800	2000	800	750	650	—
GIMBAL SYSTEM:														
BOOST PUMP:	LO_2		YES	YES	YES	YES	YES	YES	YES	YES	NO	NO	NO	
	LH_2		YES	YES	YES	YES	YES	YES	YES	YES	NO	NO	NO	
TPA:	LH_2 PUMP		2	2	3	3	2	2	3	3	1	1	1	
	LH_2 TURBINE		1	1	2	2	1	1	2	2	1	1	1	
	LO_2 PUMP		GEAR	1 1/2	1 1/2	1 1/2	GEAR	1 1/2	1 1/2	1 1/2	GEAR	1	1	
	LO_2 TURBINE		1	1	1	1	1	1	1	1	1	1	1	
PREBURNER			YES	YES	YES	YES	YES	YES	YES	YES	NO	NO	NO	
IGNITION SYSTEM			2	2	2	2	2	2	2	2	1	1	1	
ELECTRIC SYSTEM			SAME →											
INSTRUMENTATION			SAME →											
PURGE SYSTEM			SAME →											
CHILLDOWN SYSTEM			TCA BP →											
NOZZLE RETRACTION			NO	NO	NO	YES	NO	NO	NO	YES	NO	YES	YES	
ENGINE CONTROL			LV	LV	LV	LV	LV	LV	LV	LV	LV + CGV	LV + CGV	LV + CGV	
ENGINE WEIGHT		F/P	6.1	10.0	13.9	22.7	6.1	10.0	13.9	25.0	10.0	20.0	28.5	
		ϵ	450	360	270	450	450	360	270	450	300	450	600	
		W_t	180	250	340	760	180	250	340	800	200	430	710	

Figure 276. Configuration Definition

TABLE XCVII

COST IMPACT DUE TO VARIOUS DESIGN CONDITIONS

			Demonstrator Program Cost	Development Program Cost	Production 1st Unit Cost
1.	20 Vacuum Starts		$ (249,420)	$ (612,326)	$ (11,389)
2.	600 Vacuum Starts		623,550	1,530,815	28,473
3.	5.0 to 7.0 Nominal MR	5 =	37,963	93,199	1,734
		7 =	(129,074)	(316,877)	(5,894)
4.	10 Thermal Cycles (Expendable)		-	-	-
5.	60 Thermal Cycles (Reusable)		-	-	-
6.	600 Thermal Cycles		163,847	402,244	7,482
7.	2 hr Life (Reusable)		-	-	-
8.	20 hr Life (Reusable)		85,680	210,344	3,912
9.	100 to 400 (290 Max)		(1,676)	(4,115)	(77)
10.	Multiposition Nozzle		548,242	1,345,934	25,034
11.	0 Feet H_2 Pump NPSH		-	-	-
12.	15 to 60 Feet H_2 Pump NPSH		-	-	-
13.	0 Feet O_2 Pump NPSH		2,161	5,305	99
14.	2 to 16 Feet O_2 Pump NPSH		1,297	3,184	59
15.	500 sec Max Run Time		-	-	-
16.	2000 sec Max Run Time		496,300	1,219,644	22,685
17.	No Throttle Capability		(92,320)	(226,646)	(4,216)
18.	10:1 Throttle Capability		60,245	147,901	2,751
19.	2 Weeks Max Orbit Storage Time		(21,594)	(53,013)	(986)
20.	104 Weeks Max Orbit Storage Time		82,196	201,791	3,753
21.	Idle Mode		61,392	150,717	2,803
	BASELINE ENGINE		18,218,039	44,725,992	831,500

TABLE XCVIII

COST IMPACT DUE TO THRUST AND CYCLE VARIATION

	Demonstrator Program Cost	Development Program Cost	Production 1st Unit Cost
1. Staged Combustion			
A. 8K	$ (1,846,035)	$ (5,736,692)	$ (10,967,064)
B. 15K	(948,423)	(3,168,054)	(6,349,353)
C. 25K (Base for Delta Cost)	0	0	0
D. 50K	5,302,910	15,744,900	30,303,731
2. Staged Combustion Bleed			
A. 8K	(1,846,035)	(5,736,692)	(10,967,064)
B. 15K	(948,423)	(3,168,054)	(6,349,353)
C. 25K	0	0	0
D. 50K	5,518,460	16,465,513	31,746,766
3. Expander			
A. 8K	(2,297,143)	(5,319,442)	(9,893,837)
B. 15K	607,001	3,537,348	7,133,974
C. 25K	2,973,057	11,457,480	23,007,357
4. Gas Generator Cycle			
A. 8K	(2,132,404)	(5,304,634)	(10,149,964)
B. 15K	(1,277,903)	(3,144,648)	(5,820,859)
C. 25K	1,377,203	5,856,144	11,495,558
D. 50K	4,46[illegible],810	14,496,288	28,811,976

TABLE XCVIII (cont.)

		Demonstrator Program Cost	Development Program Cost	Production 1st Unit Cost
5.	TCA Tap Off			
	A. 8K	$ (2,472,665)	$ (6,183,456)	$ (11,625,478)
	B. 15K	(1,830,633)	(4,456,427)	(8,162,195)
	C. 25K	2,762,507	1,377,313	2,804,870
	D. 50K	3,692,447	12,177,492	24,450,392
6.	Coolant Tap Off			
	A. 8K	(1,501,718)	(6,903,468)	(13,068,513)
	B. 15K	470,752	(62,712)	(81,200)
	C. 25K	376,049	10,737,469	21,564,322

TABLE XCIX

25K ENGINE COST SUMMARY

	Demonstrator Program	Development Program	Production 1st Unit	Production Unit	Total Program Including 40 Units
Ground Based Reusable	18,220,000	44,730,000	1,058,000	331,000	96,200,000
Ground Based Expendable	16,400,000	40,250,000	952,000	748,000	86,570,000
Space Based Reusable	19,130,000	46,960,000	1,100,000	873,000	101,010,000

E. 10K THRUST ENGINE DESIGN

The engine design requirements for the 10K engine were slightly modified from the 25K engine design. The new requirements are presented in Table C. Table CI indicates the modified payload trade-off parameters for the 10K thrust engine.

The difference in requirements, as compared to the 25K thrust engine, are:

Minimum thrust rise rate req.

Minimum coast time req.

Zero NPSH pump assisted idle mode start

Increased engine weight payload sensitivity

Envelope definition

Of these requirements, only the latter three have an impact on the engine design. The minimum thrust rise rate and coast time requirements effect engine operation only and do not effect the basic design. Since the engine is designed for deep throttling, the start transient can be slowed to any desired rate within the throttling range. The minimum coast requirement of 60 sec has no design impact, since in this start time period heat soak back and environmental effect are minimized. However, it should be noted that repressurization of the propellant tank may be required, which may effect the design of the tank pressurization.

The pump assisted idle mode operation is not a baseline engine requirement but an alternate requirement and, therefore, is not reflected in the basic engine design. The engine modifications are identified in the following sections and the engine control requirement and engine operation are defined by steady state LETS II computer analysis which are also discussed in subsequent sections of this report.

The modified payload sensitivity parameters mean the payload is very sensitive to engine weight. Very large area ratio nozzles are feasible for the 10K engines, consequently consideration of light weight nozzles was mandatory since the nozzle weight represents the largest single component weight. The envelope constraint was modified to consist of a 400:1 nozzle but not to exceed 82-in. overall engine length. This requirement defines the nozzle as a fixed nozzle concept. The area ratio, however, must be optimized based on the payload trade-off parameters. The optimum area ratio is very sensitive to the nozzle concept (i.e., weight) and therefore careful consideration was given to concept selection.

TABLE C

10,000-POUND THRUST ENGINE OPERATING CHARACTERISTICS

Propellants	Liquid Oxygen/Liquid Hydrogen
Maximum Vacuum Thrust, pounds	10,000
Nominal Engine Mixture Ratio	6.0:1
Engine Mixture Ratio Operating Range	5.5:1 to 6.5:1
Vacuum Thrust Throttling Capability	5.0:1
Nozzle Configuration	Bell
Nozzle Expansion Ratio	400
Turbine Drive Cycle	Staged Combustion
Vacuum Specific Impulse, seconds	*
Engine System Weight, pounds	*
Number of Vacuum Starts	60
Lifetime (Expendable Mode), thermal cycles	6
Service Life Between Overhauls (Reusable Mode), thermal cycles	300
Service Life Between Overhauls (Reusable Mode), hr	10
Gimbal Angle (Square Pattern), degrees	7
Gimbal Acceleration, radians/(second)^2	20
Minimum Natural Frequency of Gimbal System, Hertz	10
Fuel Pump NPSH, feet of hydrogen	60
Oxidizer Pump NPSH, feet of oxygen	16
Maximum Single Run Duration, seconds	2000
Maximum Storage Time in Orbit (Dry), weeks	52
Maximum Time Between Firings (Coast Time), days	14
Minimum Time Between Firings (Coast Time), minutes	1
Maximum Thrust Rise Rate, lb/sec	3000
Service-Free Engine Run Time, hr	2
Service-Free Engine Firing Cycles	60

*To be Determined as a result of design and analysis.

TABLE CI

VEHICLE TRADE-OFF FACTORS

Trade-off factors based on Orbit-to-Orbit Missions:

$$\frac{\Delta PL}{\Delta Isp} = 157 \text{ lb/sec}$$

$$\frac{\Delta PL}{\Delta W_{Burnout}} = -7.4$$

1. Engine Design Point Selection

For the 10K engine design, the stage combustion cycle was ground ruled as shown in Figure 277. Within this cycle three variations were considered.

1. Stage combustion - bleed cycle
2. Gear driven LO_2 TPA
3. Independent LH_2 and LO_2 pump turbine drives

A comparison of the power balance capability was made and is presented in Figure 278 indicating the stage combustion bleed cycle to be slightly superior to the others. However, the additional specific impulse due to bleed losses of this cycle cannot be compensated for by the superior power balance capability. Therefore it was rejected in favor of the independent turbine drive. The gear driven LO_2 TPA alternate was used for an alternate engine design study.

The selection of the design chamber pressure for the 10K thrust engine was based on the thrust chamber life and power balance capability. The coolant pressure drops for a low cycle life requirements of 300 cycles, was established and defined for a 10K thrust engine as function of chamber pressure as shown in Figure 279. This figure indicates a sharp rise of coolant pressure drops and pump discharge pressure requirements with increasing chamber pressure as P_c = 1250 psia permitting a 2-stage fuel pump design.

For this selected chamber pressure, an engine power balance was made and the resulting engine cycle schematic for nominal conditions is shown in Figure 280. The basic engine cycle is identical to the 25K thrust engine cycle, the main difference is the reduction of the fuel pump stages by one.

An analysis was made to investigate the payload sensitivity to the assumed chamber pressure. Figure 281 indicates the relationship of engine performance and engine weight as function of chamber pressures and nozzle area. In this figure, lines of constant payload and constant engine length are shown and is based on the parameter engine study. Of interest is the fact that for constant engine length an increase of chamber pressure of 250 psia results in a payload increase of 400 lb. The payload sensitivity to chamber pressure for constant engine length is:

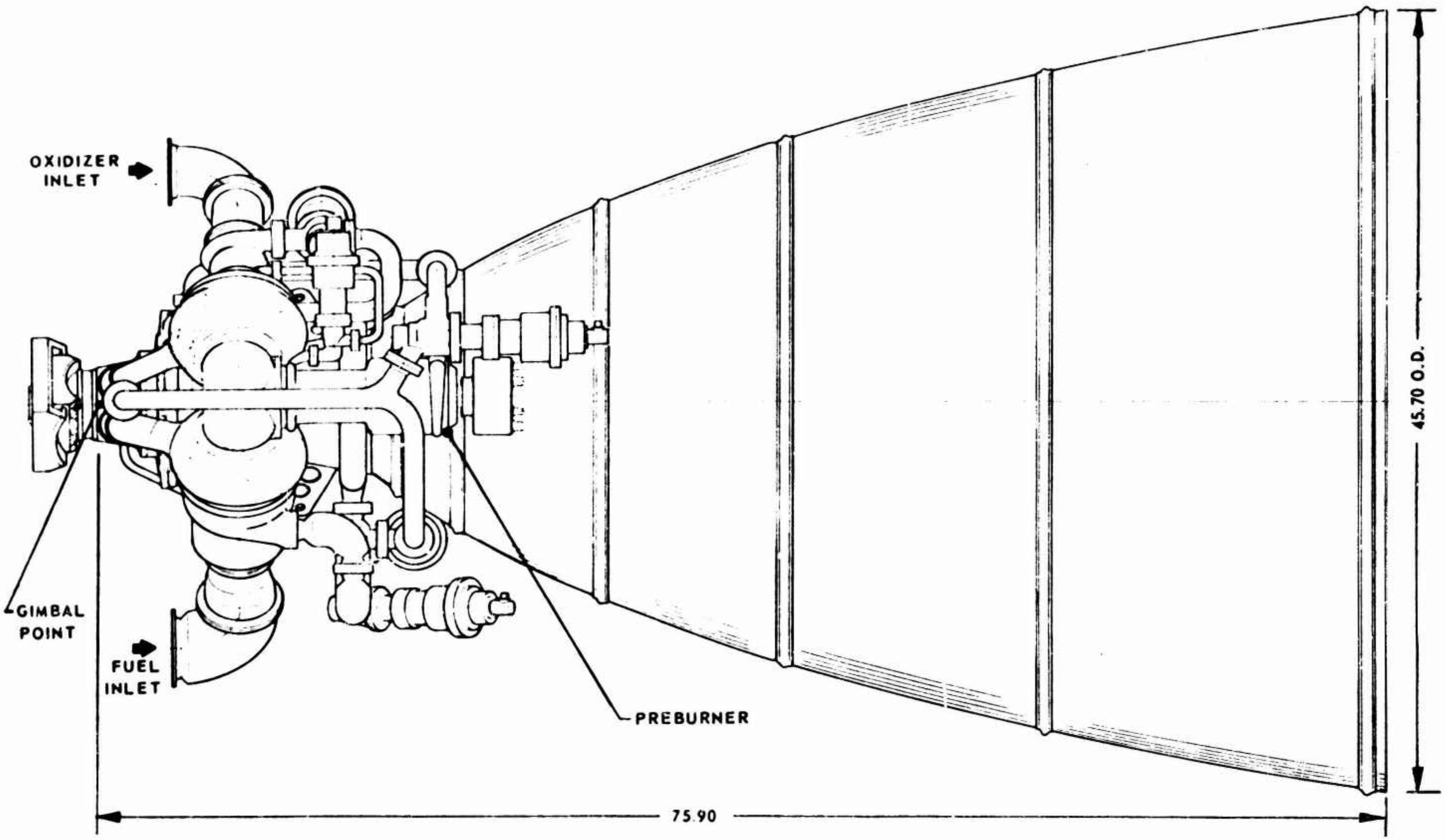


Figure 277. 10K Engine Baseline Engine Configuration

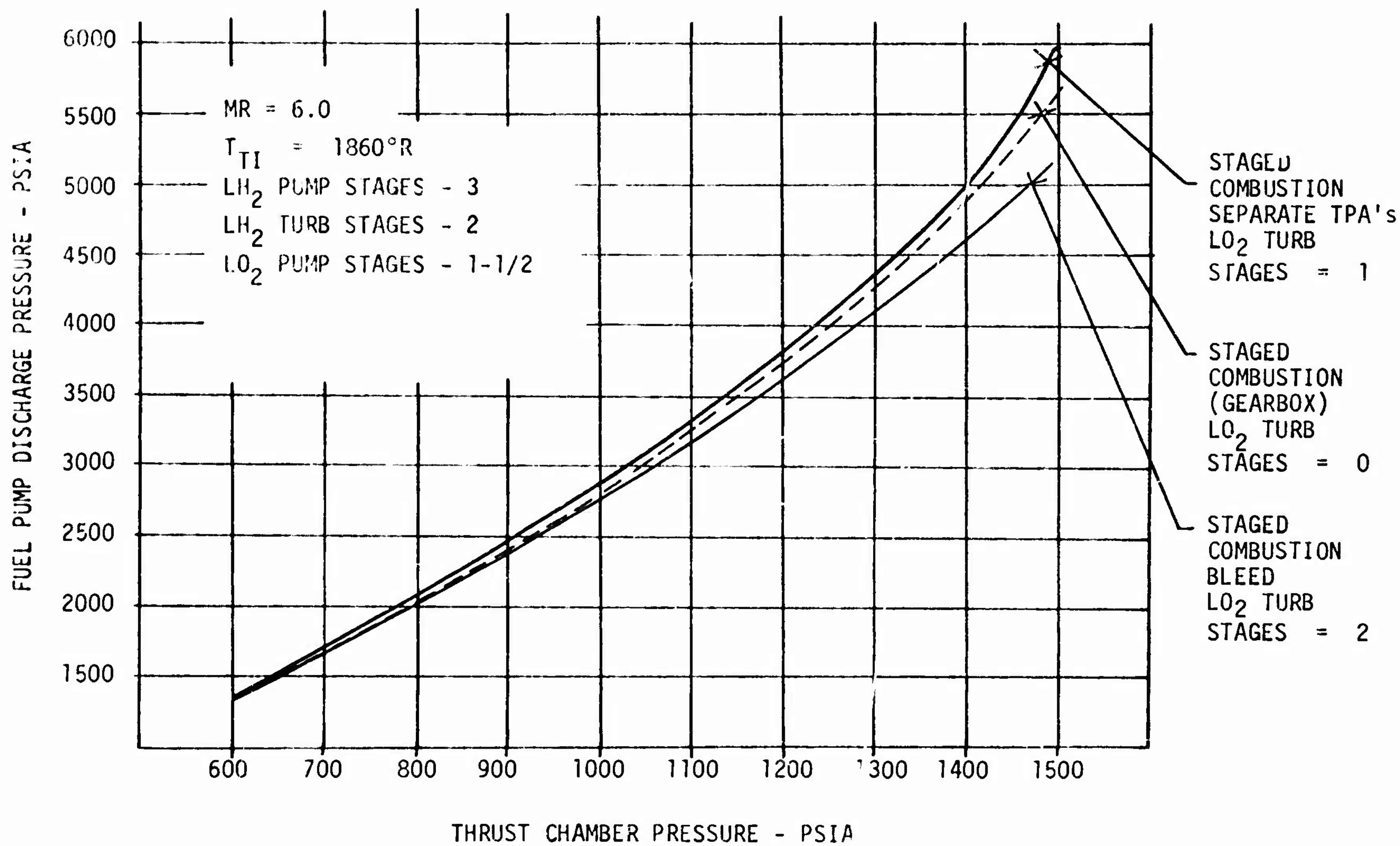


Figure 276. OOS 10K Engine Power Balance Analysis

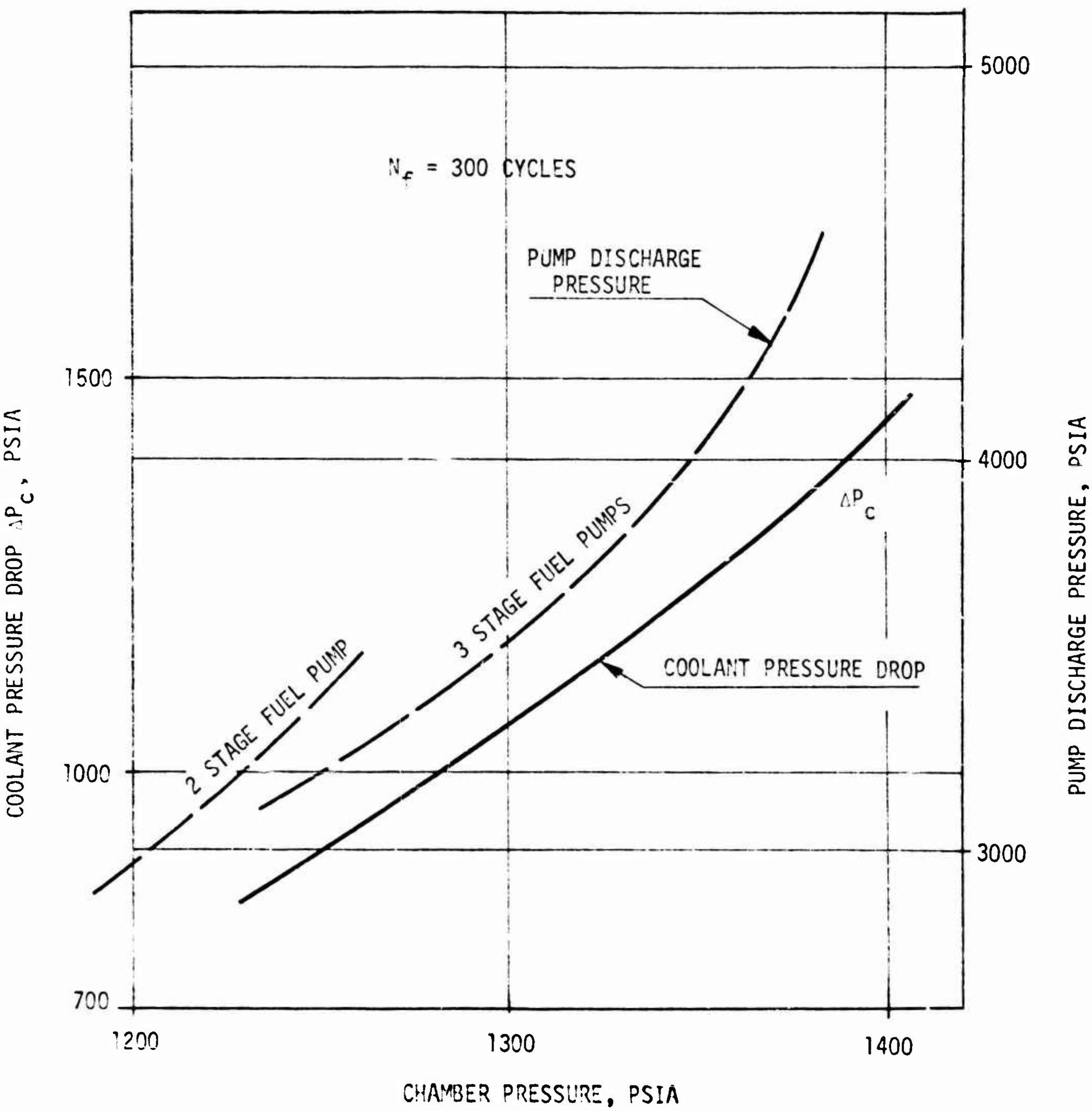


Figure 279. Coolant Pressure Drop and Pump Discharge Requirements

F = 10K LBS, MRE = 6.0 NOMINAL

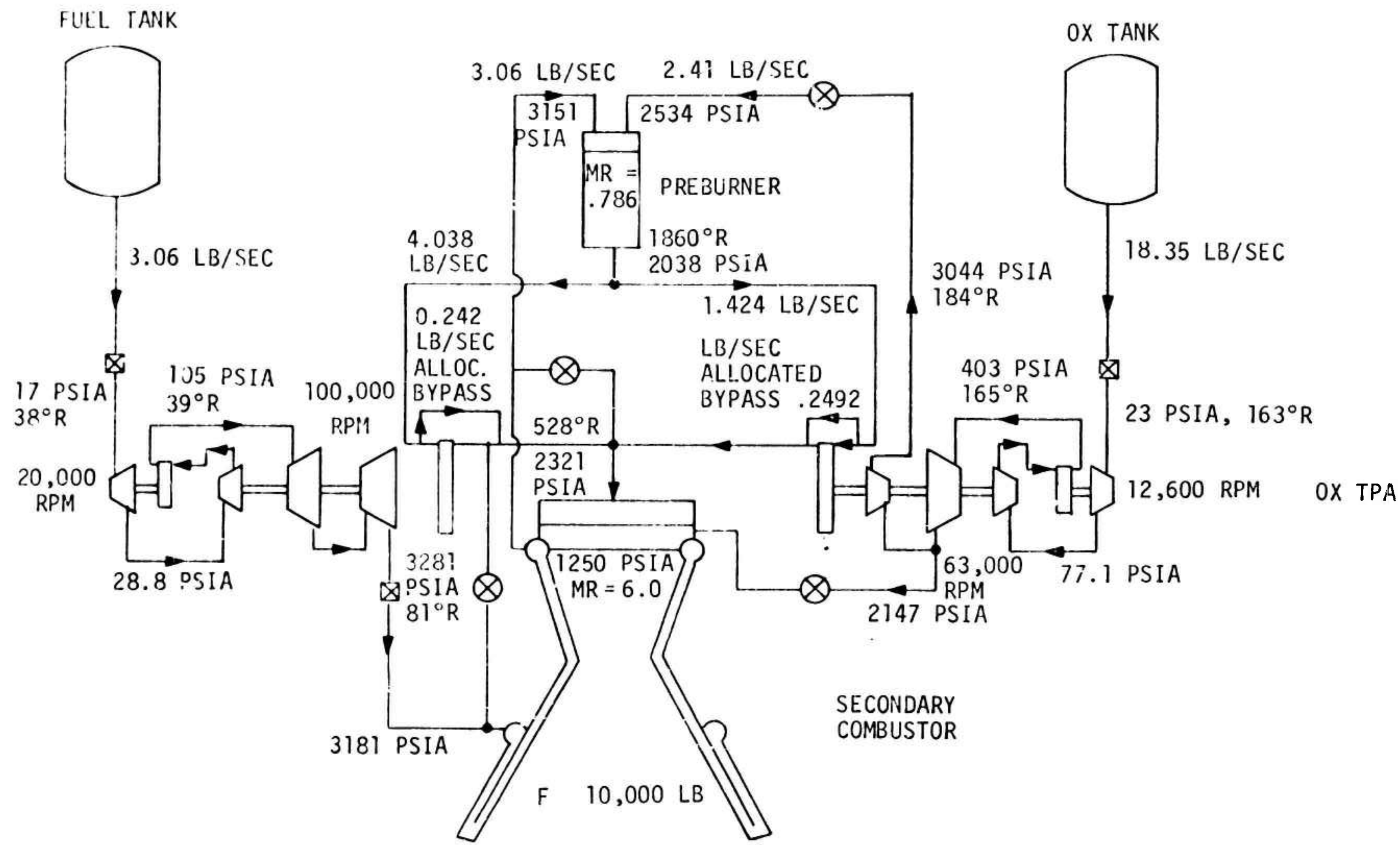


Figure 280. Engine Cycle Schematic of Nominal Conditions

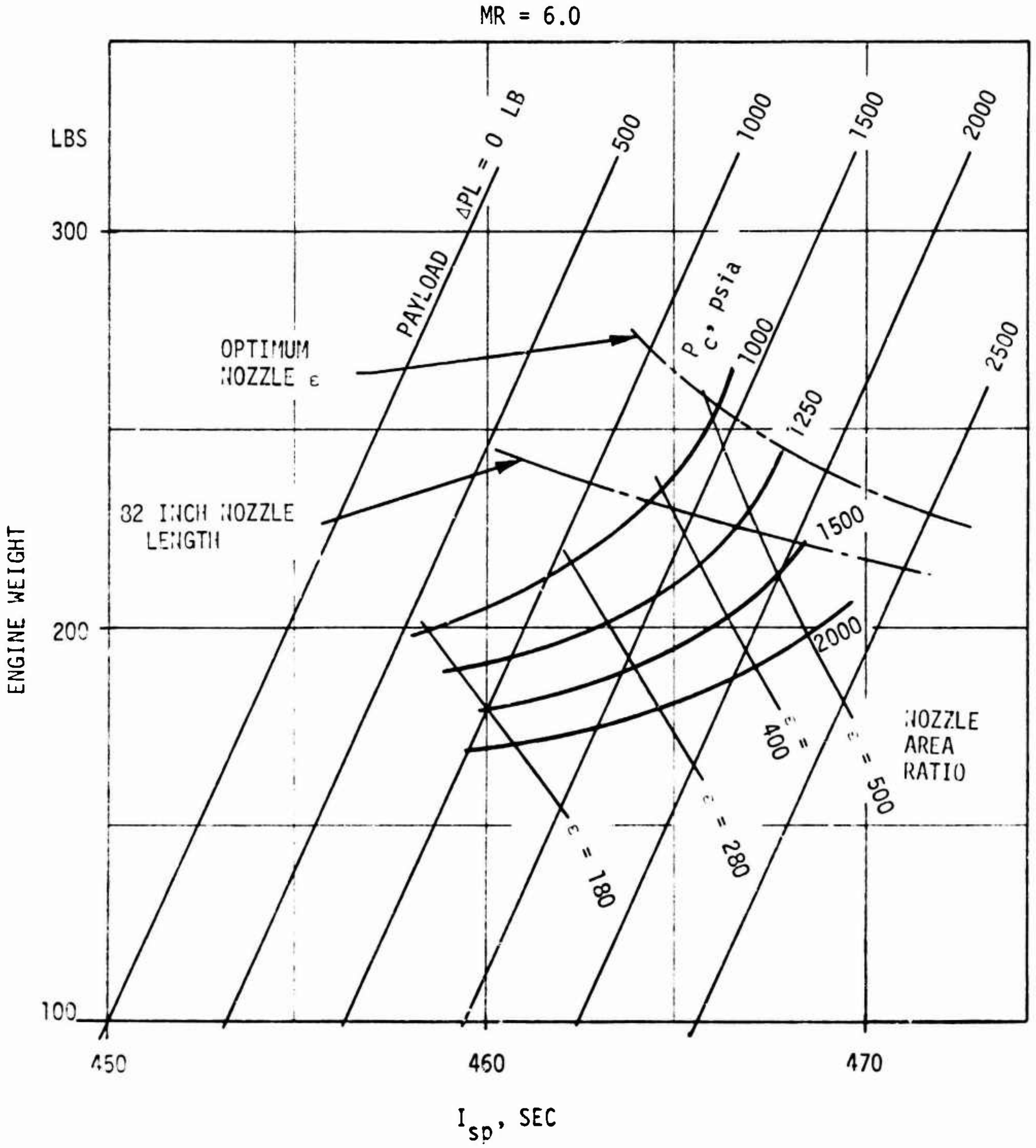


Figure 231. Payload Sensitivity Analysis

10K Thrust: $$\left.\frac{\Delta P_L}{\Delta P_c}\right|_{L=\text{Const.}} = 1.6 \text{ lb/psia}$$

25K Thrust: $$\left.\frac{\Delta P_L}{\Delta P_c}\right|_{L=\text{Const.}} = 1.2 \text{ lb/psia}$$

The selection of chamber pressure at the 10K thrust level is considerably more critical with respect to payload than for the 25K engine. Based on this, a re-evaluation of the latest thrust chamber material test data was conducted (Section III.F.10). It indicates that chamber life is very sensitive to environmental conditions effecting chamber life by an order of magnitude, and the basic life data assumed was conservative.

2. Nozzle Expansion Area Ratio Selection

The relatively low thrust 10K engine has the capability of a large nozzle expansion area ratio within the engine length constraints of 82 in.

For the nozzle contour the minimum length Rao nozzle was used and the nozzle characteristics such as engine length, diameter and performance were determined as function of area ratio.

Figure 282 presents the JANNAF calculated performance as function of area ratio and chamber pressure for the 10K thrust staged combustion engine. The performance is calculated for an energy release efficiency of 99%. The performance shown in Figure 282 is considered conservative, since the interim JANNAF performance prediction method yields 1-1/2 sec higher specific impulse. Test experience shows that at these chamber pressures ERE = 99.5% can be achieved. Figure 283 represents the performance tolerance band estimate for the 10K thrust engine, indicating that performance of 470 sec, of specific impulse can be achieved.

The engine overall length relationship as function of expansion area ratio is shown in Figure 284 indicating that the maximum feasible nozzle expansion area ratio within 82-in. engine length is ϵ = 460:1 resulting in a maximum engine diameter of D_{max} = 49 inches and a performance potential of I_s = 466.5 to 470.4 sec.

The design expansion area ratio was optimized for the given engine payload sensitivities. The very large area ratios and the large engine weight sensitivity factors makes the design area ratio strongly dependent on the nozzle configuration.

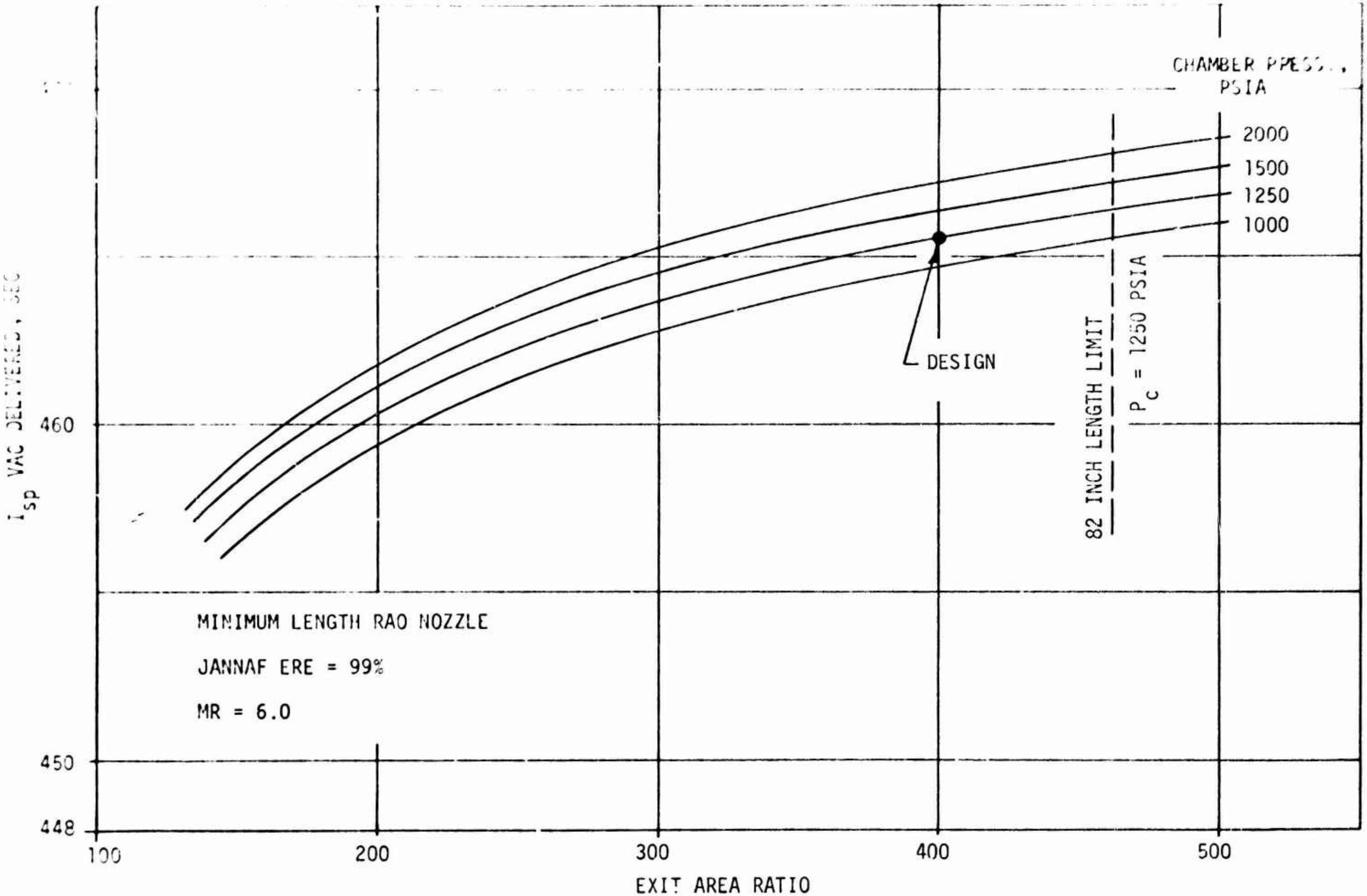


Figure 282. 10K Engine Performance as a Function of Area Ratio and P_c

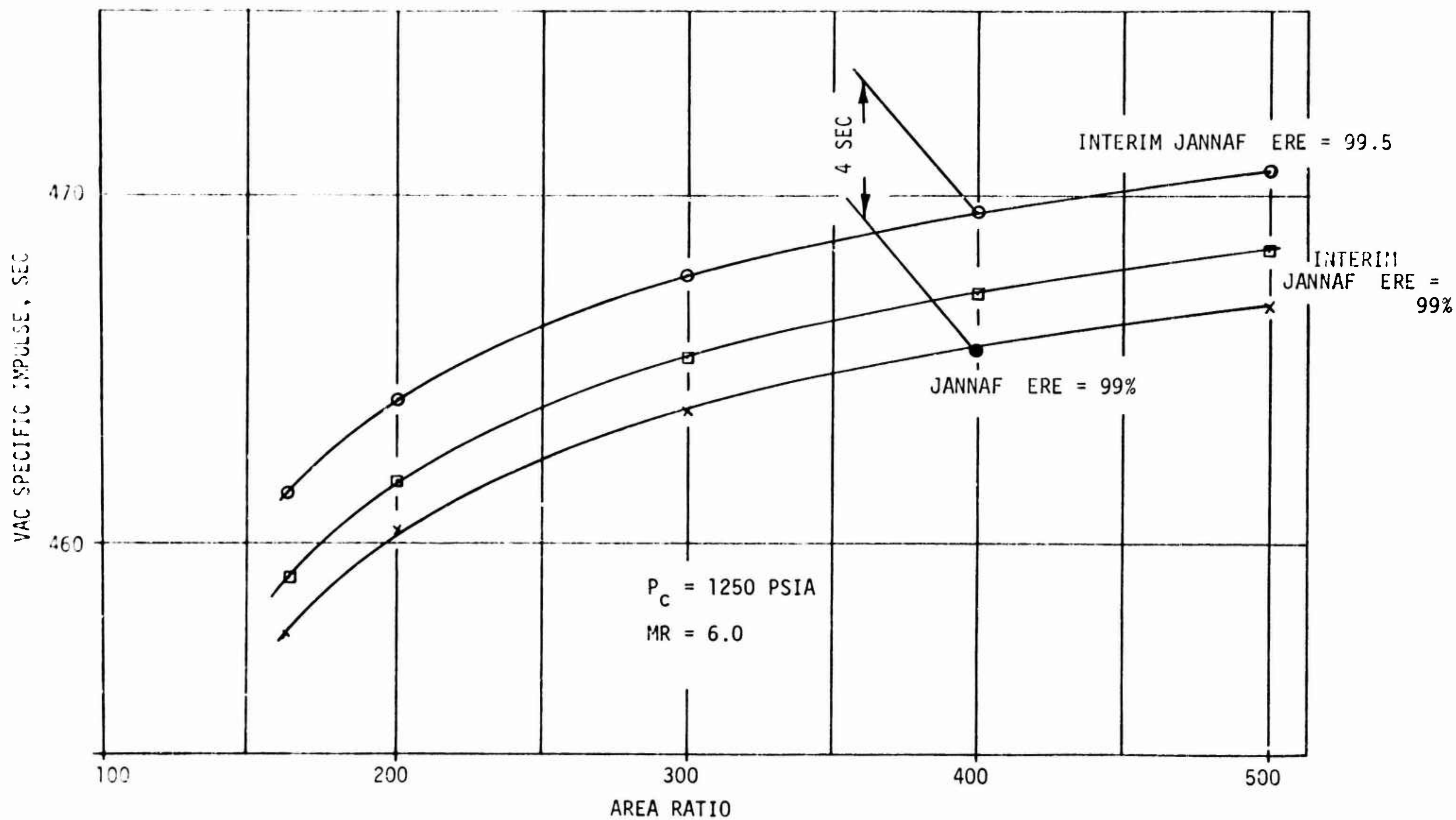


Figure 283. 10K Engine Performance Potential

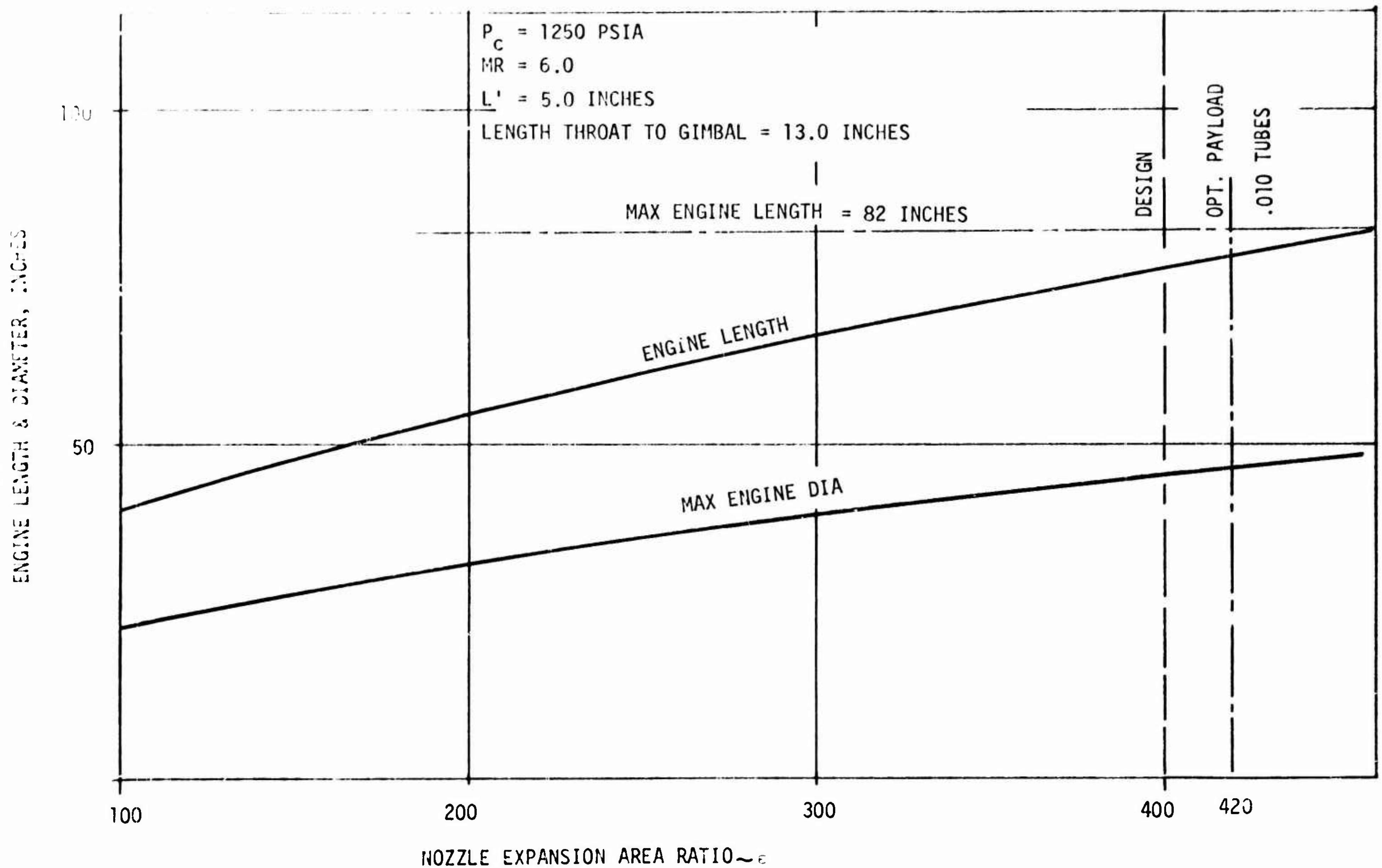


Figure 284. Engine Overall Length as a Function of Expansion Area Ratio

An analysis was conducted for two different types of nozzle extensions:

All regeneratively cooled

Combination regeneratively and radiation cooled

For the all regeneratively cooled nozzle, the transition area ratio from the machined copper chamber to the tubular nozzle extension is ϵ = 5.5:1. The nozzle weight of the tubular section was established as function of expansion area ratio and tube wall thickness of 0.15 in. and 0.010 in. and is presented in Figure 285.

For the radiation cooled nozzle AGCarb material of 0.125 in. wall thickness was assumed. The effect of the transition area ratio from regenerative cooling to radiation cooling on nozzle weight was analyzed and is presented in Figure 286, for an overall expansion area ratio of ϵ = 400:1.

As shown, the transition area ratio has a significant effect on the radiation cooled nozzle weight. Heat transfer analysis indicates that the transition area ratio of ϵ = 70:1 is the lowest area ratio possible for uncoated graphite resulting in a total nozzle weight of 64 lb. Further reduction of this area ratio would require Hafnium coating of the nozzle to obtain the desired life capability versus nozzle erosion. Such a nozzle would permit attachment at the area ratio of 5.1:1 and would reduce the nozzle weight to 46 lb.

The maximum attachment point of a radiation cooled nozzle is ϵ = 150:1. At this area ratio, the radiation cooled configuration is equal to the all regeneratively cooled nozzle.

To establish the effects of nozzle configuration and payload capability, a nozzle area ratio optimization study was made. The results are shown in Figure 287.

Figure 287 relates nozzle extension weight versus engine performance for the all regeneratively cooled nozzle, the regenerating and radiation cooled nozzle, and the all radiation cooled nozzle. The results indicate that the all regeneratively cooled nozzles reach an optimum payload limit within the 82 in. engine length; the radiation cooled nozzles however are length limited. The payload changes due to nozzle configuration are significant:

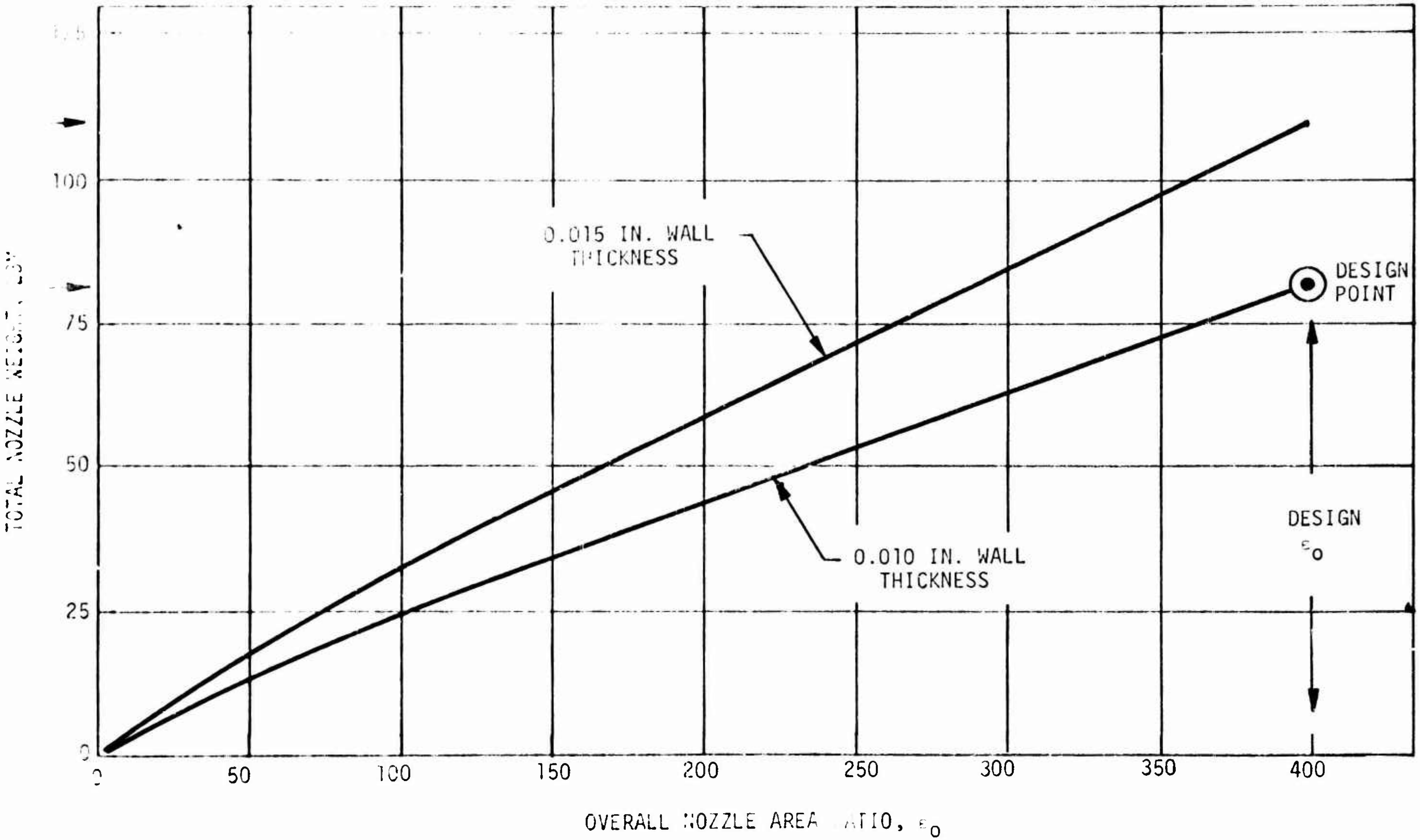


Figure 285. Nozzle Weight vs Overall Area Ratio

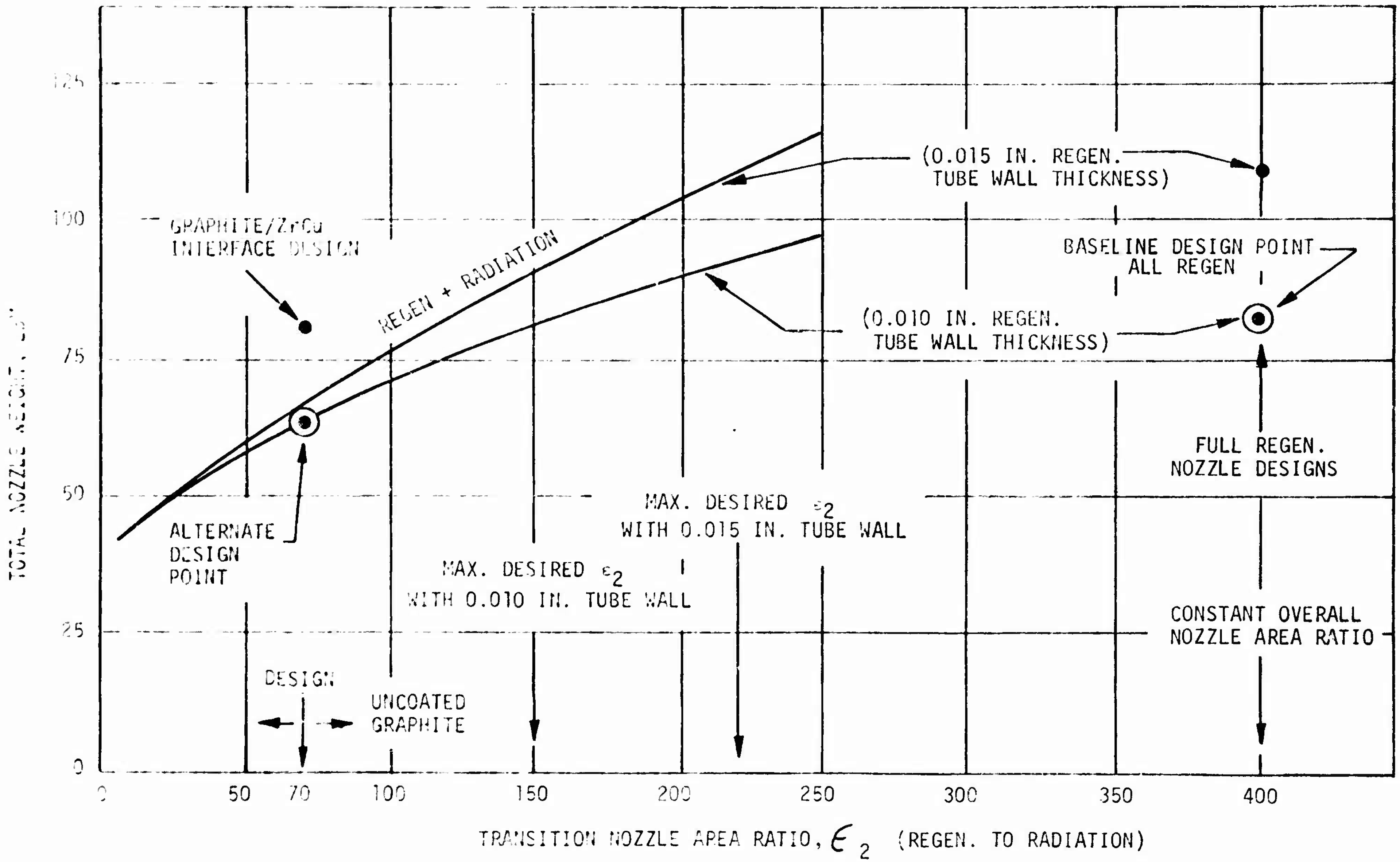


Figure 286. Nozzle Weight vs Transition Area Ratio for Regen and Radiation Cooled Nozzles

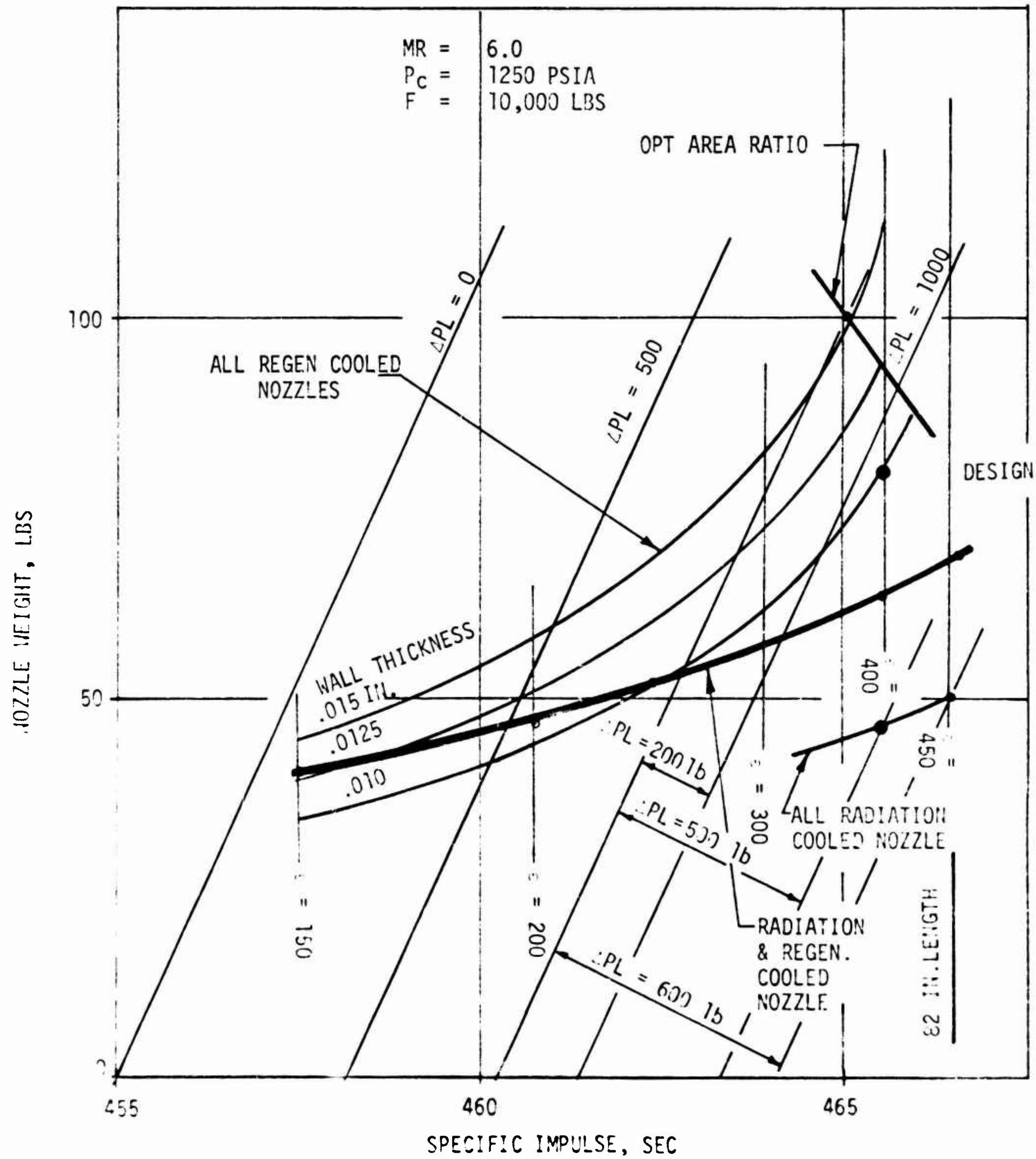


Figure 287. Nozzle Wall Thickness vs Payload

Nozzle Concept Comparison

Configuration	Material Thickness in.	Max Area Ratio	Nozzle Weight lb	I_s sec	Length in.	ΔP_L Payload lb
All Regeneratively Cooled Nozzle	0.010	440	104	465.9	80	0
	0.0125	400	95	465.6	76.5	-125
	0.0150	370	88	465.0	74.0	-250
Regeneratively and Radiation Cooled Nozzle $\varepsilon_o = 70:1$	0.125 (0.010)	460	69	466.5	82	+250 lb
All Radiation Cooled Nozzle $\varepsilon_o = 5.5:1$	0.125	460	50	466.5	82	+350 lb

As shown, the total payload span due to nozzle configuration is ΔP_L = 600 lb. As the baseline configuration, the all regeneratively cooled chamber was assumed with a tube wall thickness of 0.010 in. and the selected area ratio is ε = 400. The reduction of the area ratio to ε = 400 from the optimum ε_{opt} = 440 has no effect on payload capability according to Figure 287.

The loss in payload due to decreased area ratio is shown in Figure 288 and was calculated based on the given sensitivity factors. At ε_a = 300:1 the engine length would be 68 inches at an I_{sp} = 464 and an engine weight change of ΔW_E = 17.5 lb as compared to ε = 400:1 resulting in a payload loss of 150 lb.

Further advanced technology nozzles have the potential to increase payload by 250 to 350 lb as compared to the selected baseline.

The selection of the regeneratively cooled nozzle was made on the basis of readily available technology. However the selected tube wall thickness of 0.010 in. is considered the absolute minimum and would be of high fabrication cost as compared to a 0.015 in. tube wall. In addition, it is anticipated that in a reusable system, the 0.010 tube may prove to be too sensitive for handling and result in high maintenance cost.

The all radiation cooled nozzle would not only be sturdier, result in considerable payload gain, but also permit engine sea level testing under actual operating conditions, since it would be seperable at area ratio ε = 5.5.

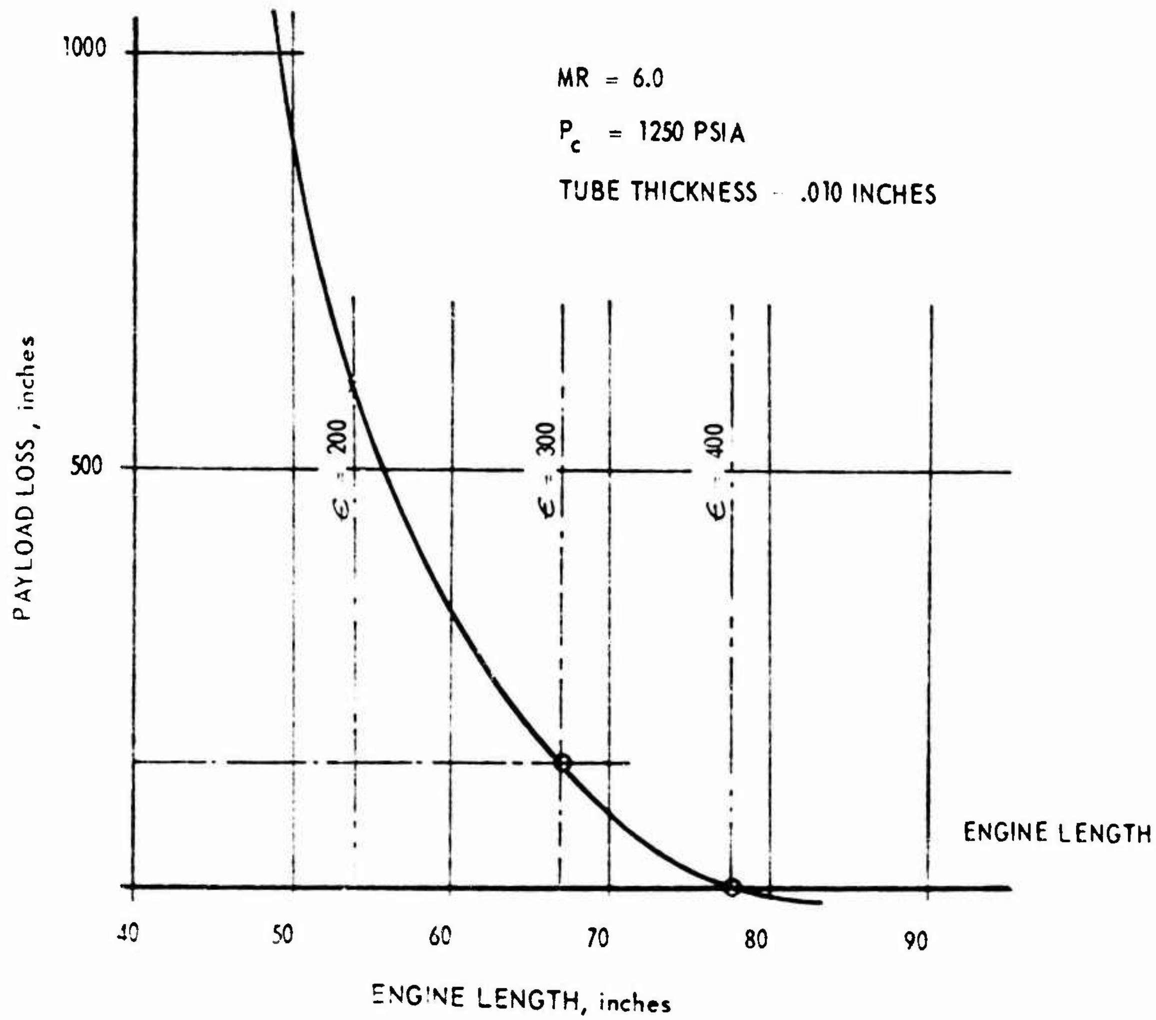


Figure 288. Engine Length vs Payload Loss

3. Basic Engine Cycle Description

a. Nominal

The engine cycle was established ~~for~~ the following nominal engine design point:

Thrust	=	10,000 lb
Chamber Pressure	=	1250 psia
Nozzle Area Ratio	=	400:1
Mixture Ratio	=	6.1
Turbine Inlet Temperature	=	1860°R
Specific Impulse	=	465.5 sec (469.5 I_s max)

The flow schedule presented in Figure 280 is representative of the selected engine design point. The engine cycle is designed to permit a throttle range of 5:1 over the required mixture ratio range of MR = 5.5 to MR = 6.5 and is identical to the 25K thrust selected engine cycle.

For the engine control, the three valve concept was established, permitting the engine to operate at pressure fed idle mode. The engine throttle performance was established and is shown in Figure 289 indicating a fast drop-off in performance beyond 30% of thrust indicating that engine throttling beyond this point is not desirable.

b. Off Design Engine Operation

A LETS 2 model of the baseline 10K engine was set up and operated over a throttling range of 10:1 and for mixture ratios of 5.5, 6.0, and 6.5. The results are summarized in plot form in Figures 290 through 298.

The control system used was essentially the same as that on the 25K engine, oxidizer preburner and thrust chamber valves were used for control of thrust and mixture ratio and the fuel preburner bypass valve was used to maintain a constant turbine inlet temperature of 1750°R below 75% thrust.

Results were very similar to those obtained for the 25K engine. The turbine temperature in the controlled range was increased from 1660°R in the 25K engine to 1760°R in the 10K engine because of the somewhat lower preburner mixture ratios (for a given temperature) resulting from higher fuel bulk temperatures leaving the cooling jacket.

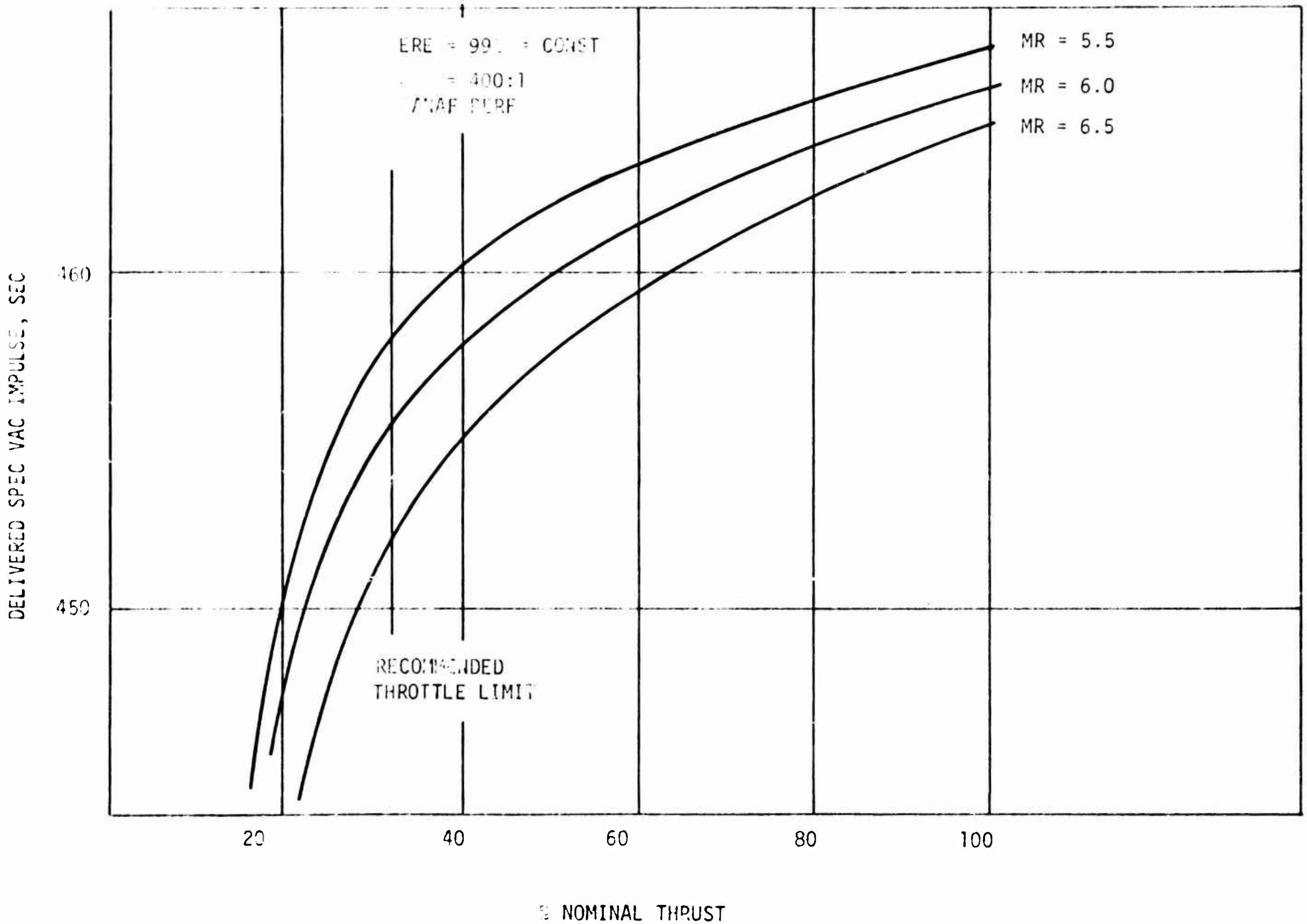


Figure 289. Throttled Engine Performance

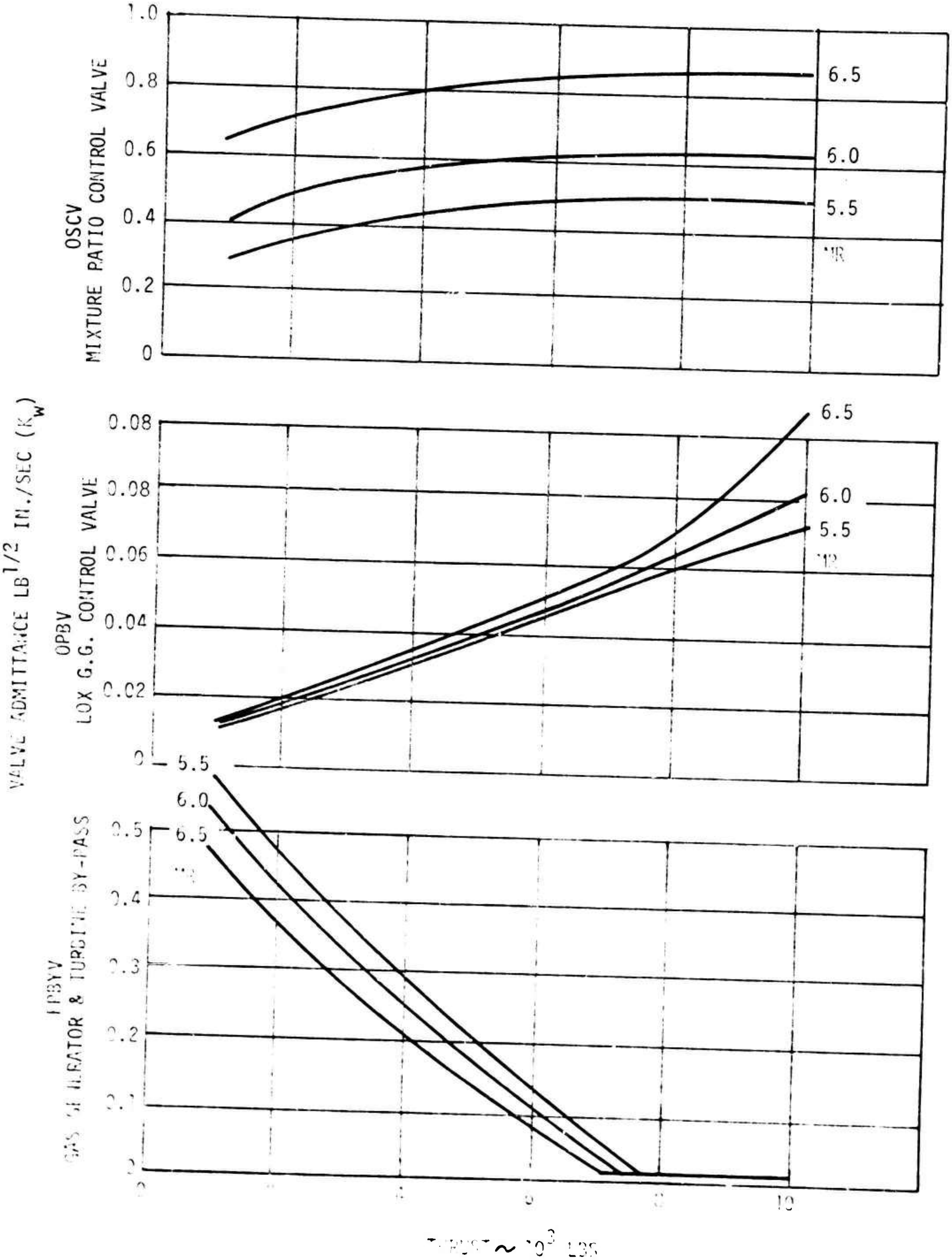


Figure 299. Control Valve Admittance vs Thrust and Mixture Ratio

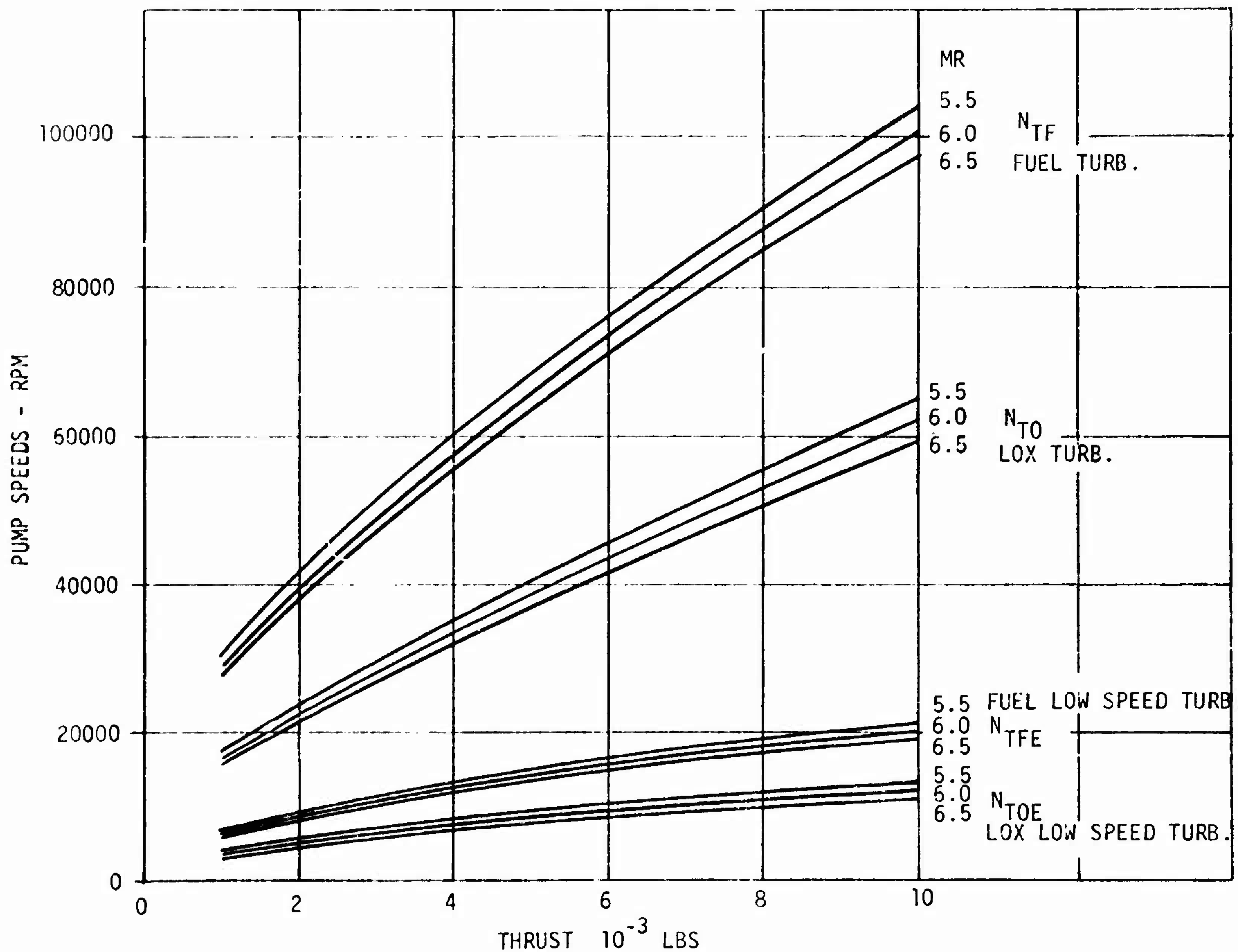


Figure 291. Turbine Speed vs Thrust and Mixture Ratio

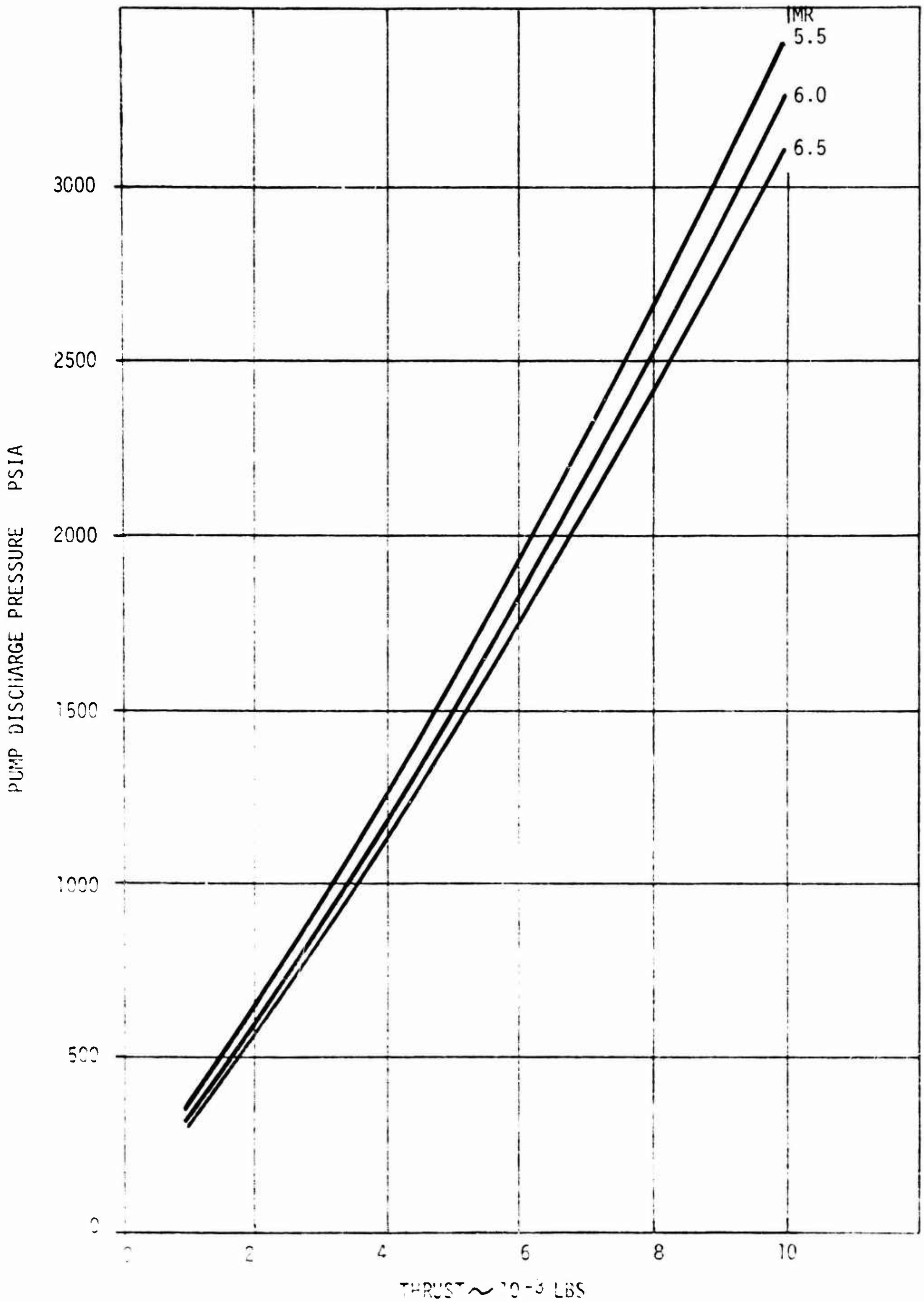


Figure 292. Fuel Pump Discharge Pressure vs Thrust and Mixture Ratio

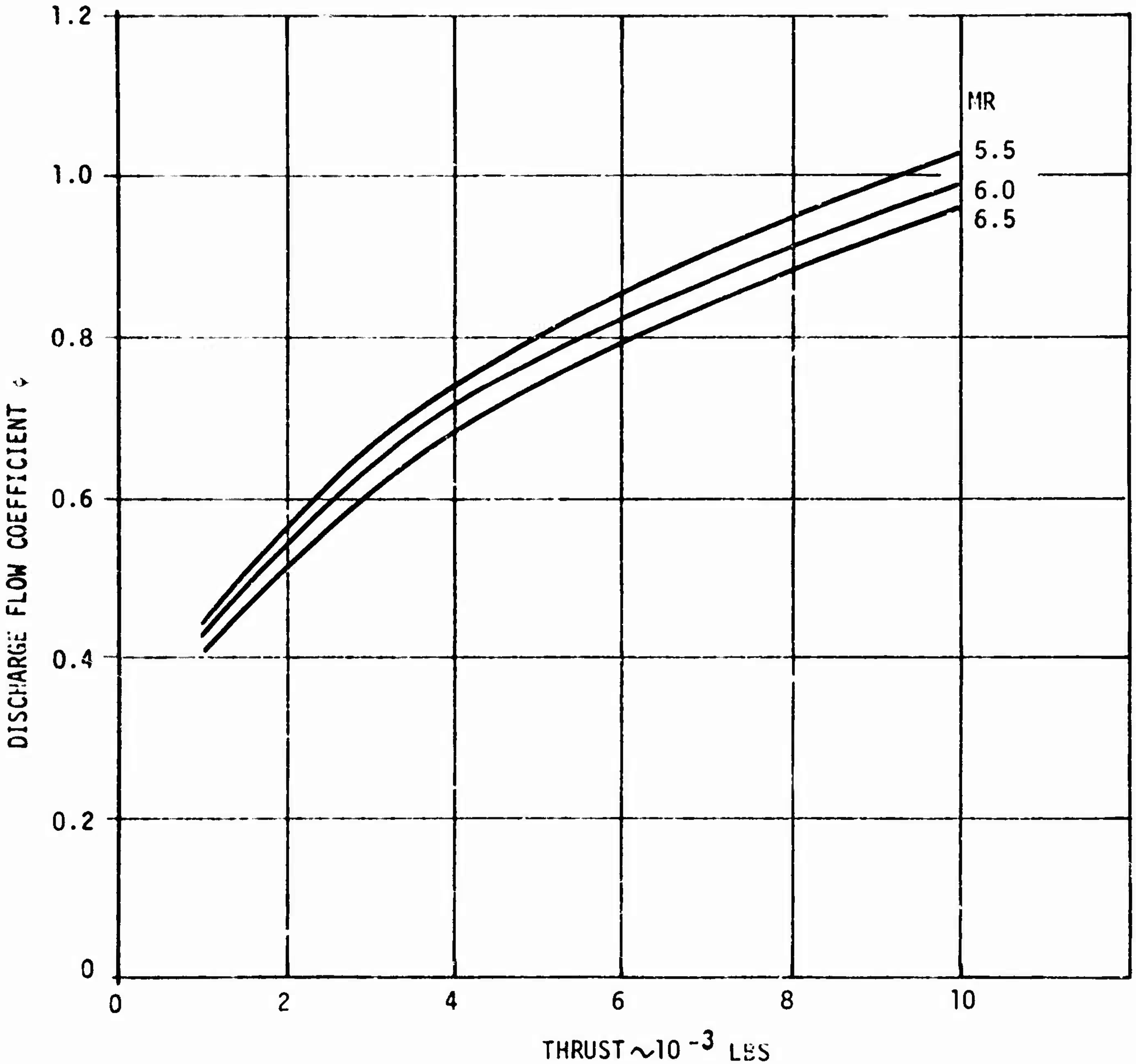


Figure 293. Flow Coefficient-Fuel Pump vs Thrust and Mixture Ratio

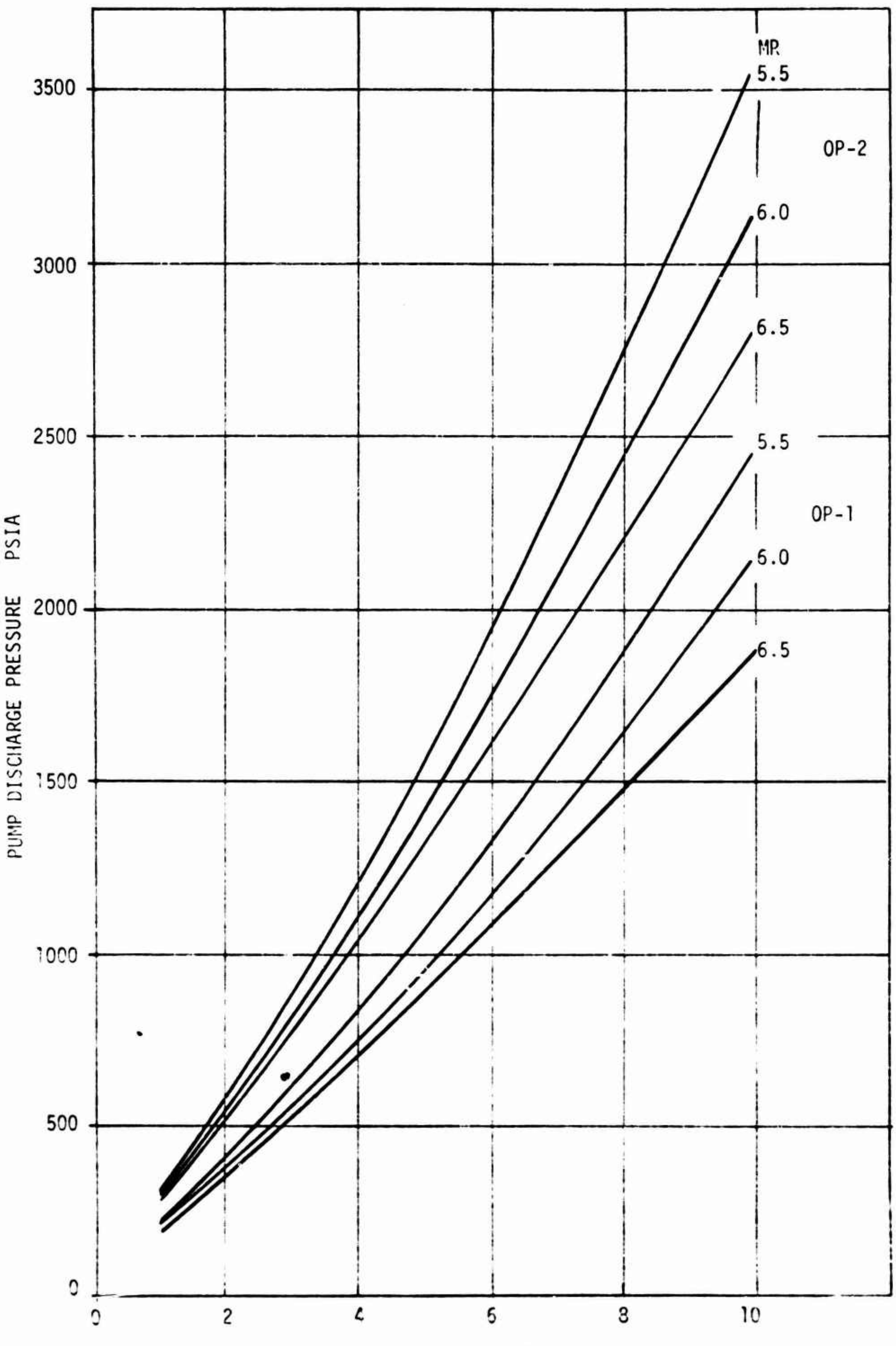


Figure 294. Main and Half-Stage Oxidizer Pumps Discharge Pressure vs Thrust and Mixture Ratio

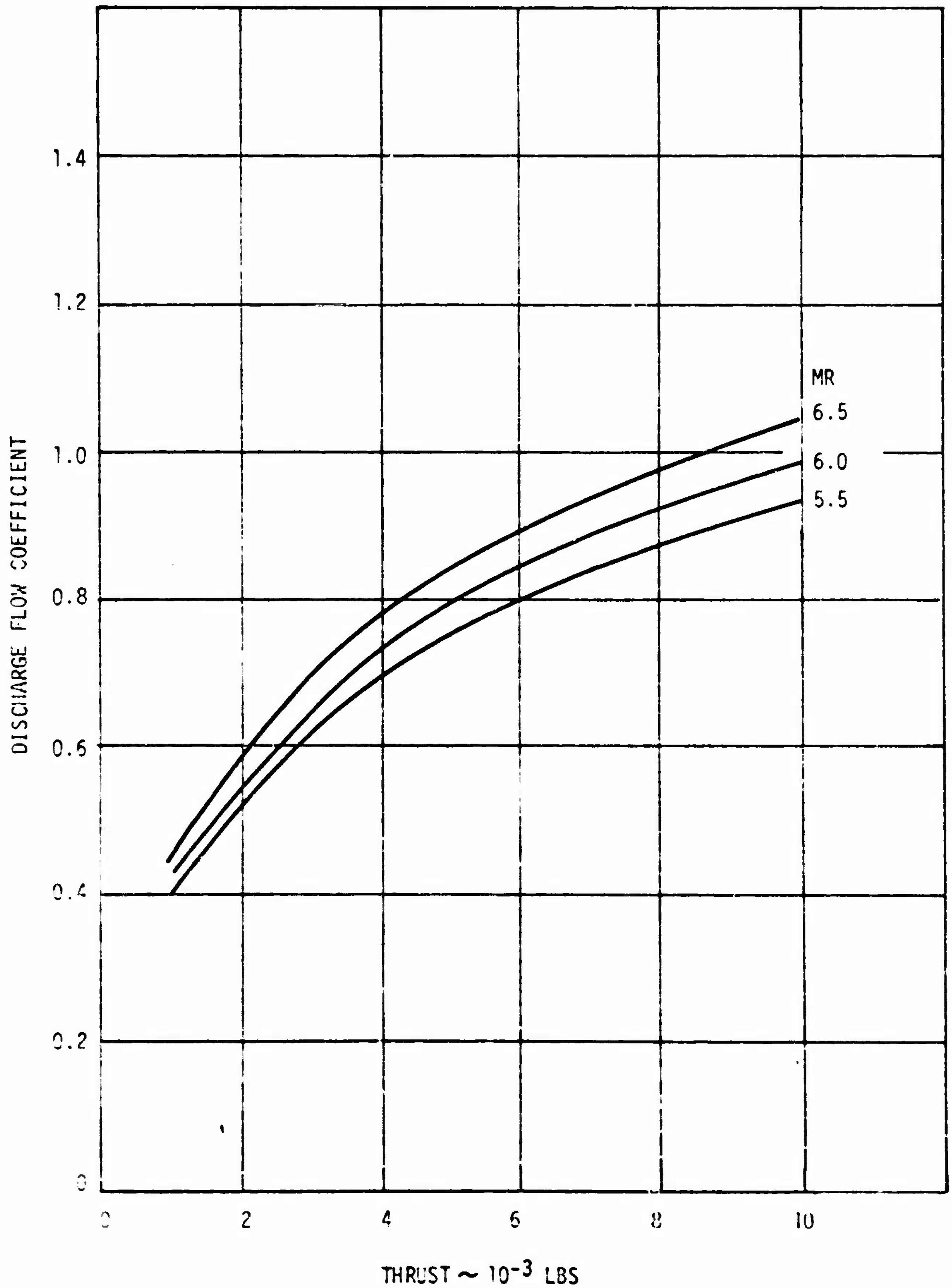


Figure 295. Flow Coefficient-Main Oxidizer Pump vs Thrust and Mixture Ratio

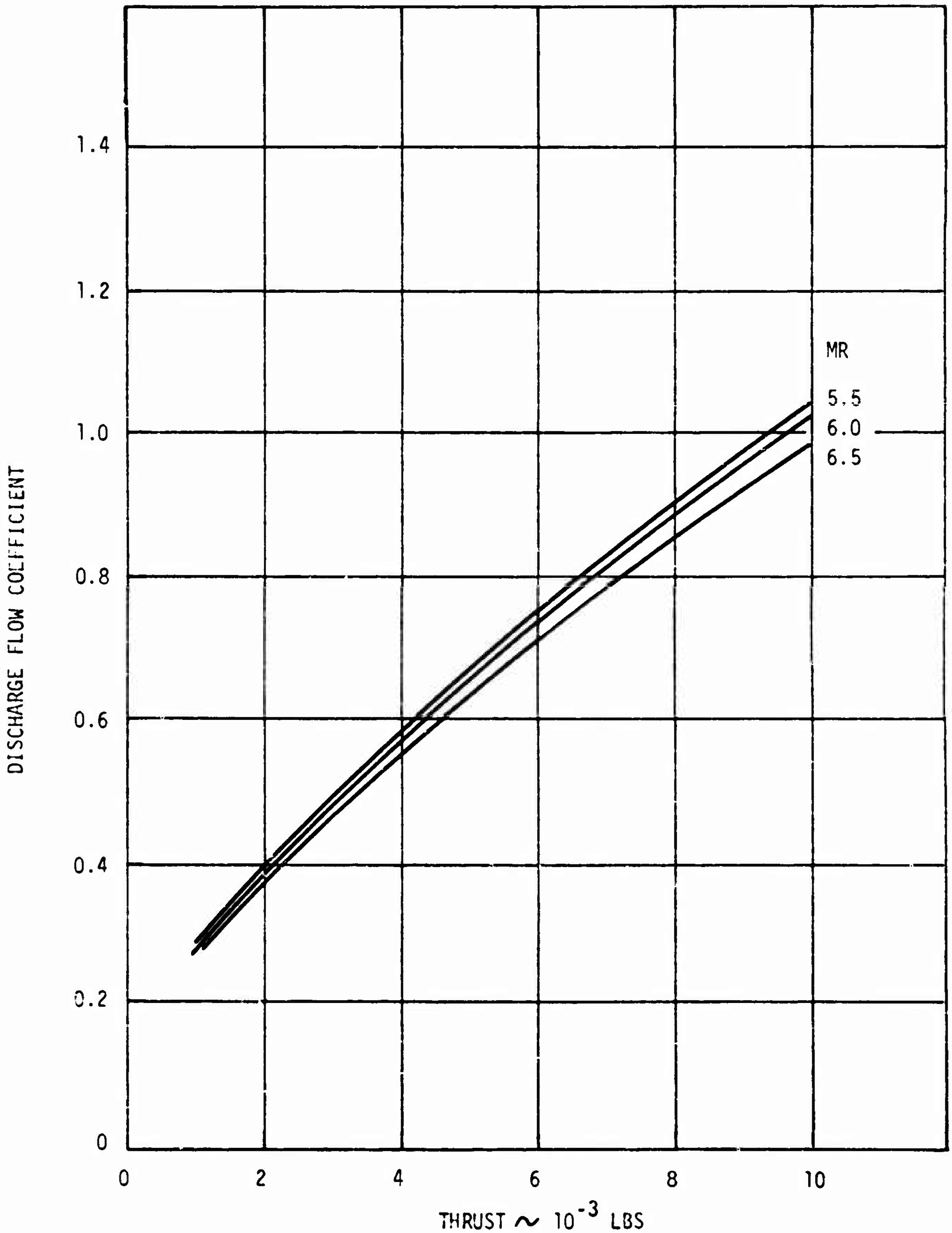


Figure 296. Flow Coefficient-Half-Stage Oxidizer Pump vs Thrust and Mixture Ratio

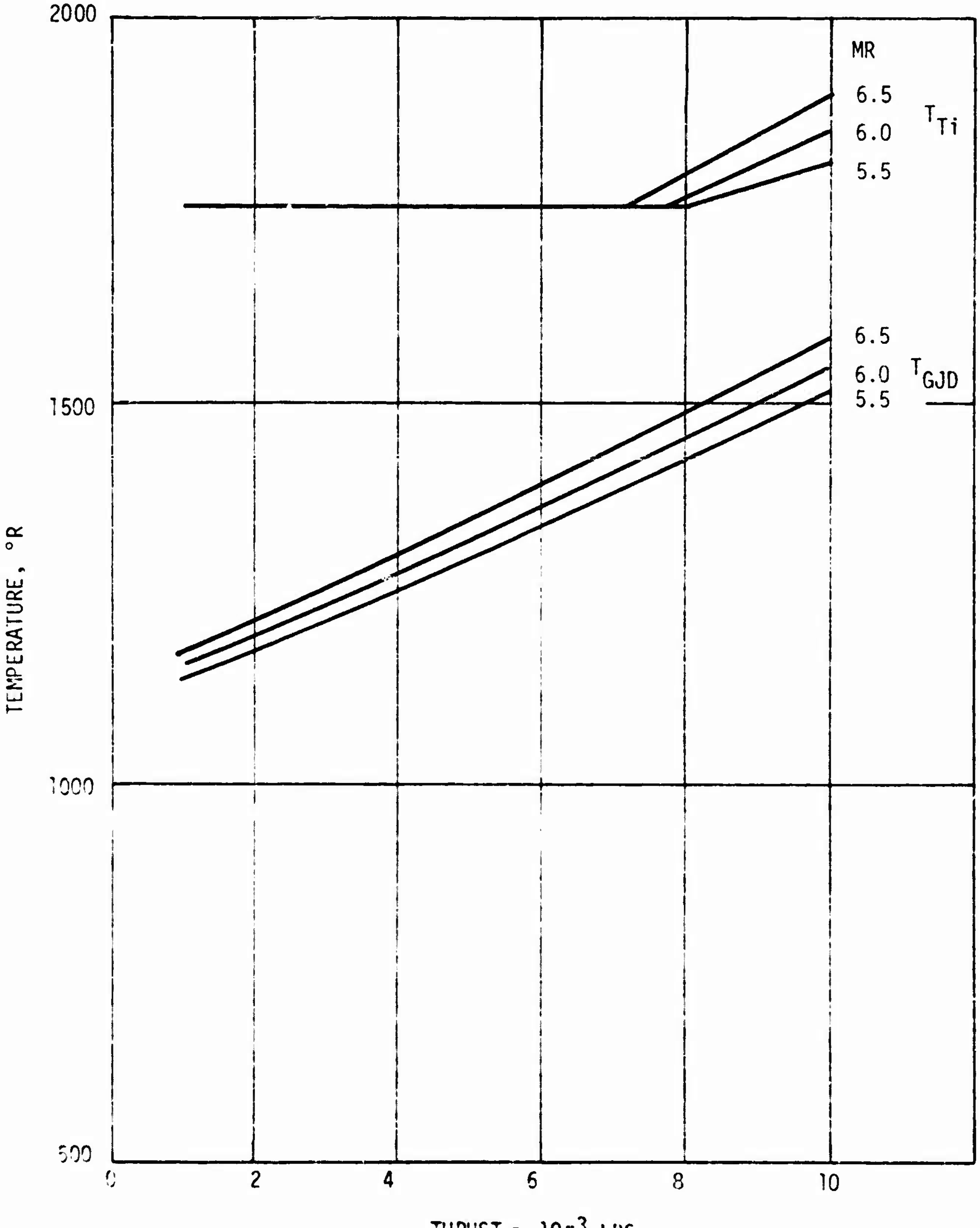


Figure 297. Turbine Inlet Temperature and Gas Injector vs Thrust and Mixture Ratio

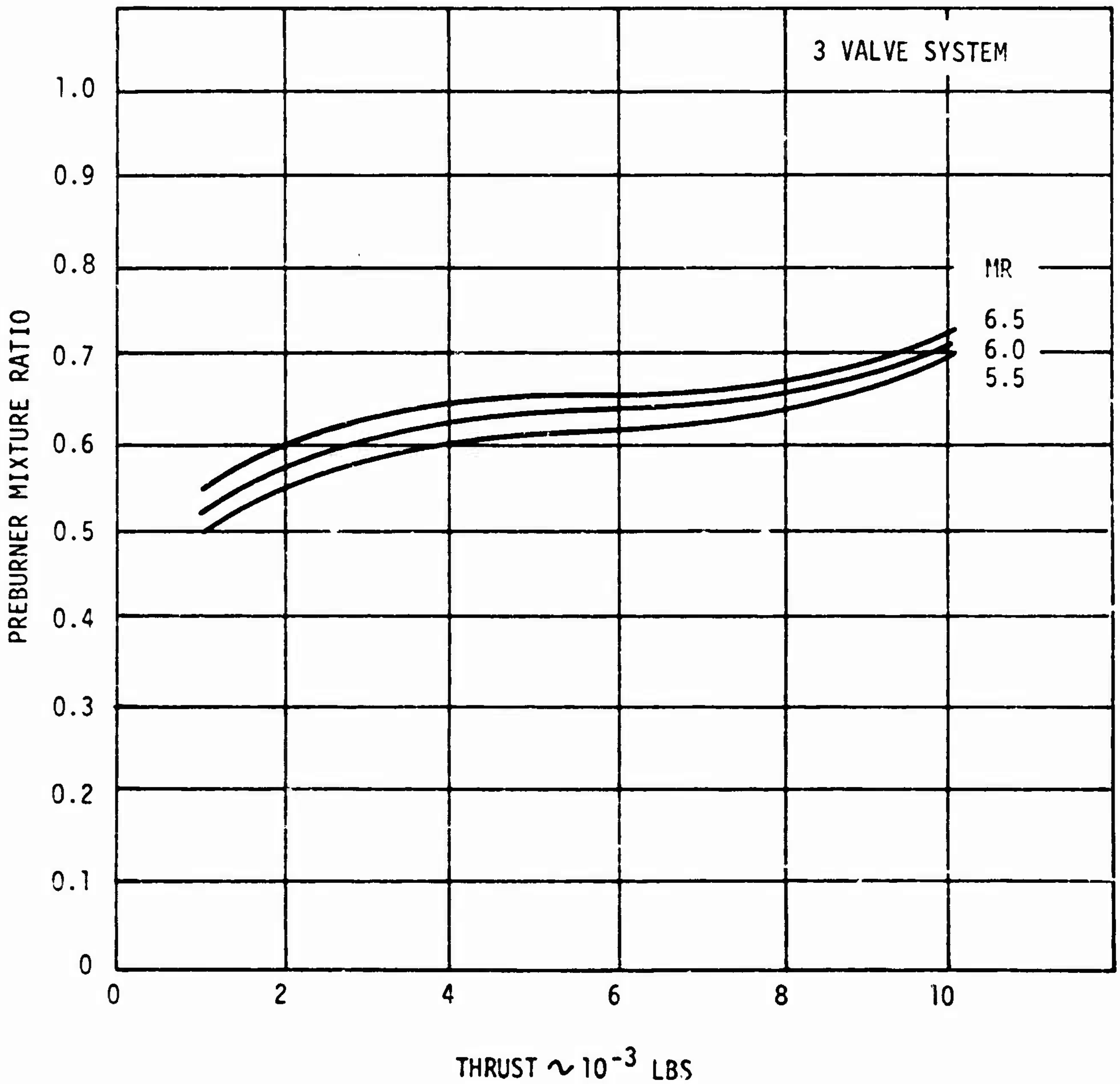


Figure 298. Preburner Mixture Ratio vs Thrust and Mixture Ratio

The engine throttle characteristic was also evaluated for its effect on the engine life and thermal cycle capability in particular for the sensitive components such as the thrust chamber and turbine disks and bearings.

For the thrust chamber thermal cycle life capability two characteristic parameters were identified, the hot side wall temperature T and the thermal wall gradient ΔT_W. The criteria of the selections of the design chamber pressure and coolant mach numbers was based on these two parameters and are shown for design conditions in Figure 321 (Section III, E, 9) as functions of chamber length. A heat transfer study was conducted to evaluate the effect on low cycle fatigue life due to off-design engine operation.

For the throttled condition the pertinent parameters are shown in Figure 319 (Section III, E, 9), indicating a rapid increase in low cycle fatigue life N_T at the reduced thrust levels. The turbine inlet temperature and speed are also reduced during throttled engine operation (Figure 297 and Figure 291) effecting turbine disk life and bearing life favorably. The results of the analysis indicate, that engine throttling is very beneficial to engine life and low cycle fatigue capability.

This fact can be utilized to prolong the engine life through engine operating procedures. Full engine thrust is only required during the initial phase of the first burn. For the subsequent burns the thrust level is not significant and throttled operation is feasible within engine performance constraints.

The engine delivered specific impulse requirements are only critical for a few large ΔV missions and many missions can be accomplished at derated specific impulse such as throttled engine performance. Even with the prolonged engine burn time requirements of throttled operation, the turbine disk and bearing life in terms of number of missions are considerably increased as compared to full thrust operation. The low cycle fatigue life of the chamber is not affected by burn time.

Therefore engine life and mission capability can be improved by mode of operation if the engine is capable of throttling.

The increased capability can be explained by either reducing engine overhaul and maintenance cost or increased payload capability through increased chamber pressure and turbine temperature for the critical missions.

4. 10K Thrust Engine Configuration

The configuration of the engine components for the 10K thrust engine are very similar to the 25K engine components. A packaging study was performed to investigate the feasibility of in-line fuel pump for the 10K configuration.

Since the selected area ratio results in an overall engine length 6 in. shorter than the available envelope (if the engine is a scaled version of the 25K configuration) the in line fuel pump concept should result in a relative small payload loss and would eliminate the complex hot gas manifold and lines of the side mounted configuration. A preliminary payload investigation shows:

Payload analysis for in-line and side mounted configuration

	In-Line	Side Mounted
Overall Length	82	76
Injector face to gimbal	16 in.	6 in.
Gimbal to Injector	13 in.	3 in.
Nozzle Area Ratio	356:1	400:1
I_s	465.0	465.5
Δ Payload (Nozzle Only)	-29 lb	0
Wt Nozzle	64 lb	72 lb

The payload loss due to loss in performance however should be compensated by eliminating the hot gas manifold because of the magnitude of the engine weight trade-off factor.

The hot gas manifold weighs 55 lb of which approximately 45.7 lb can be saved with an in line design resulting in a net gain of payload.

$$\Delta P_L = 7.4 \times 45.7 - 29 = +303 \text{ lb}$$

Due to this reasoning, the 10K engine configuration is shown as both with an in-line fuel pump and with side mounted pumps.

Although the in-line engine configuration could be designed for the same basic engine cycle as described for the baseline engine. An alternate engine cycle was studied. In this cycle the LO_2 pump is driven by a reduction gear, (designed to transmit 282 horsepower) eliminating the hot gas manifold completely.

The advantage of such a system is the rigid coupling of the two pump speeds and relative ease of engine start transient control. The reduction gear will be cooled by hydrogen gas and is considered state-of-the-art technology. A schematic of the gear driven concept including pressures and temperatures is shown in Figure 299.

The alternate engine cycle potential is also a stage combustion cycle and will achieve the same specific impulse as the baseline engine. The engine control method selected is identical to the baseline concept.

The engine configuration for both configurations are presented in Figure 300 through Figure 302 and the characteristics for both configurations are shown in Table CII and Table CIII.

Which configuration will ultimately be selected depends largely on the available engine envelope. Should the available engine length decrease considerably, then the side mounted configuration appears attractive.

The side mounted TPA appears to provide easier access to the pumps and pump may be changed without dismounting the engines. Without a stage configuration, however, this apparent advantage is difficult to evaluate.

5. Idle Mode Operation

Idle mode operation for the 10K thrust engine design is studied in two different modes:

a. Pressure Fed Idle Mode

The purpose of this operating mode is to chill both pumps simultaneously prior to starting and also to recover some specific impulse of the childown propellants.

In this operating mode, the propellants pass from the tank through the pumps into the thrust chamber. The turbines are not operating since the flow is bypassed and no pressure rise is obtained in the pumps.

It is assumed that both propellant tanks are settled at initiation of the idle mode. The tank pressures at initiation of the idle mode are assumed to be propellant vapor pressure.

No modification of the engine is required to accommodate the pressure fed idle mode since turbine preburner by-pass valve and lines are already incorporated in the baseline engine configuration. Thrust obtainable from this mode of operation will depend on the minimum tank pressure available.

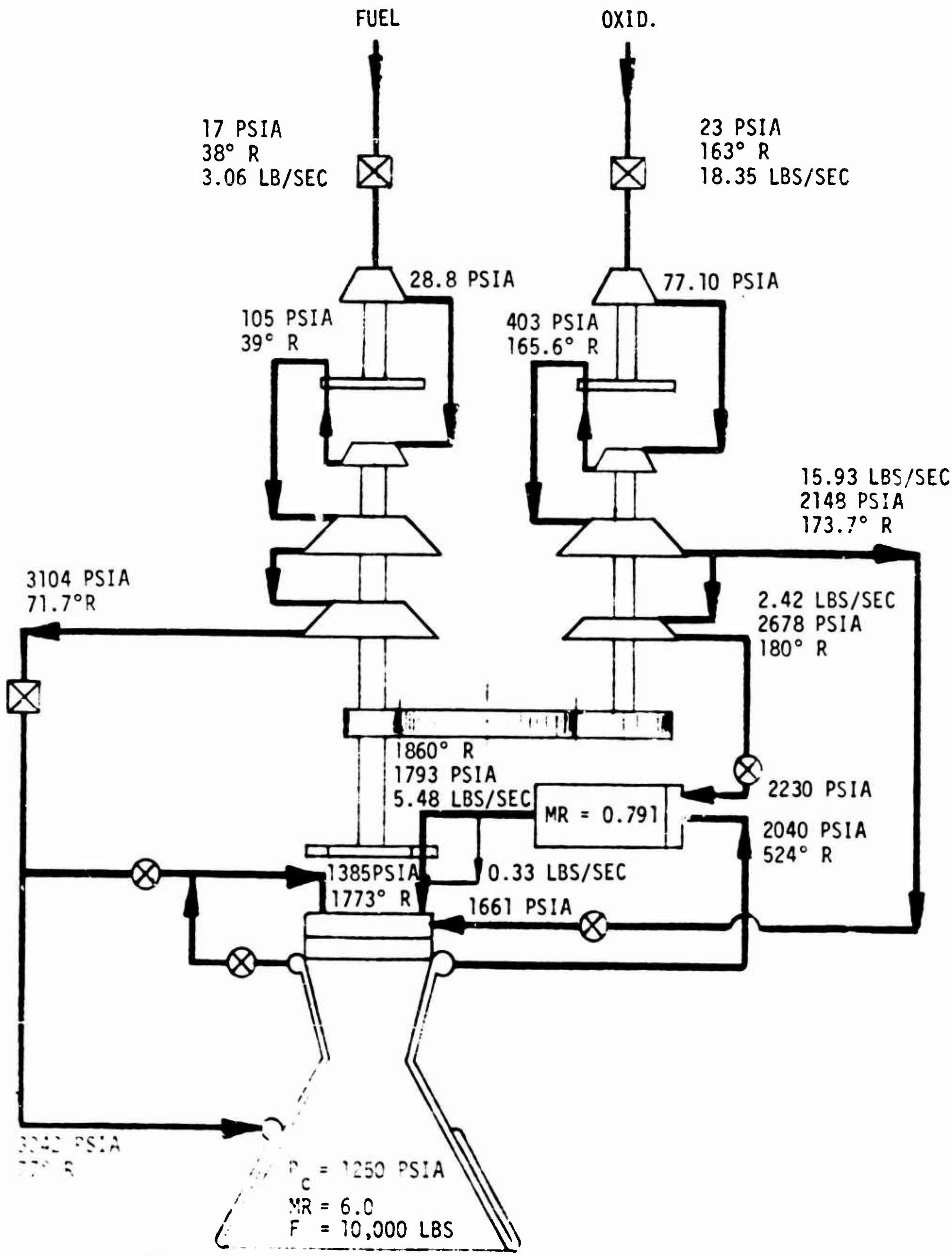


Figure 299. Engine Schematic, Gear Driven LO_2 Pump

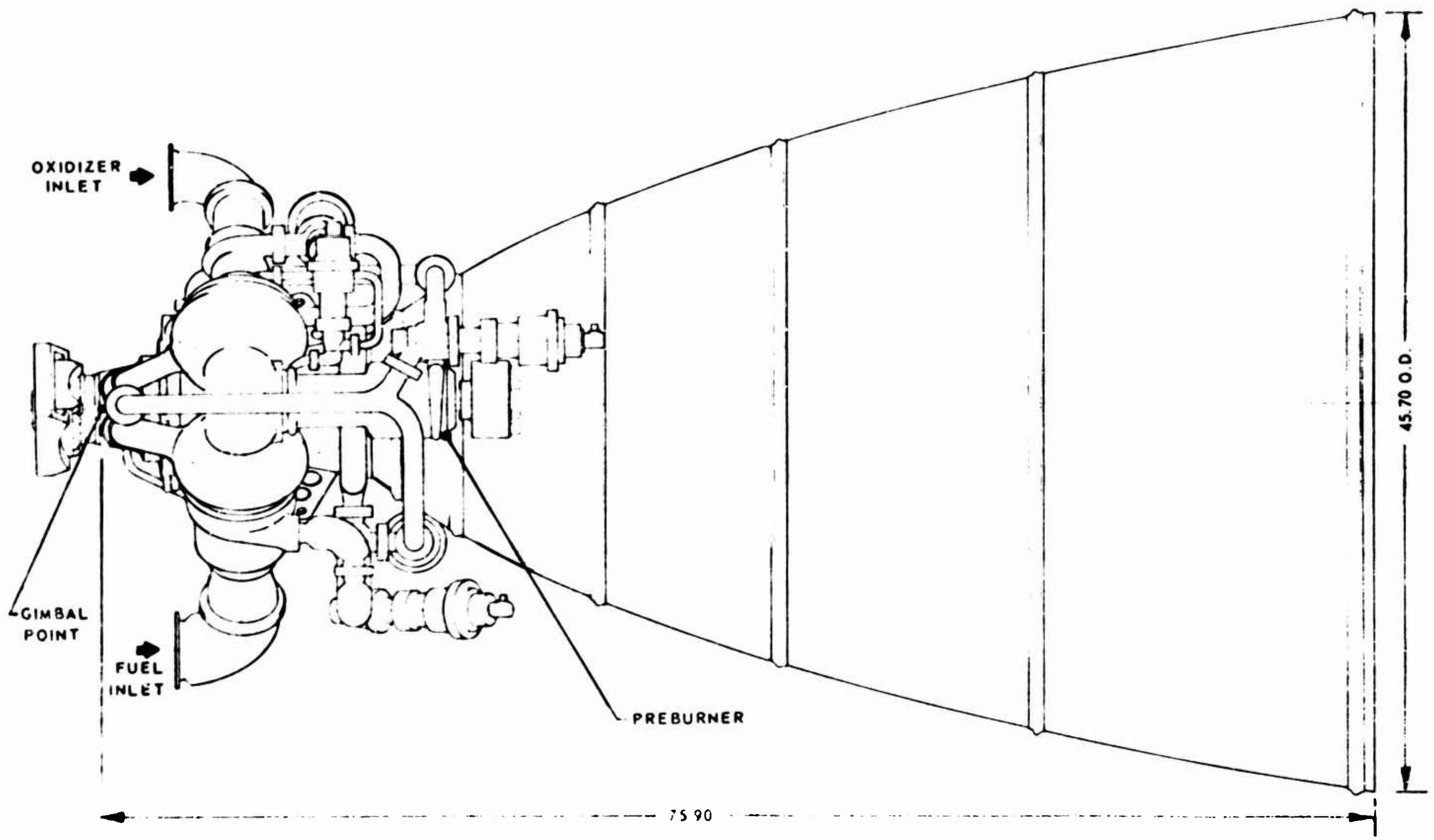


Figure 300. 10K Engine Baseline Engine Configuration

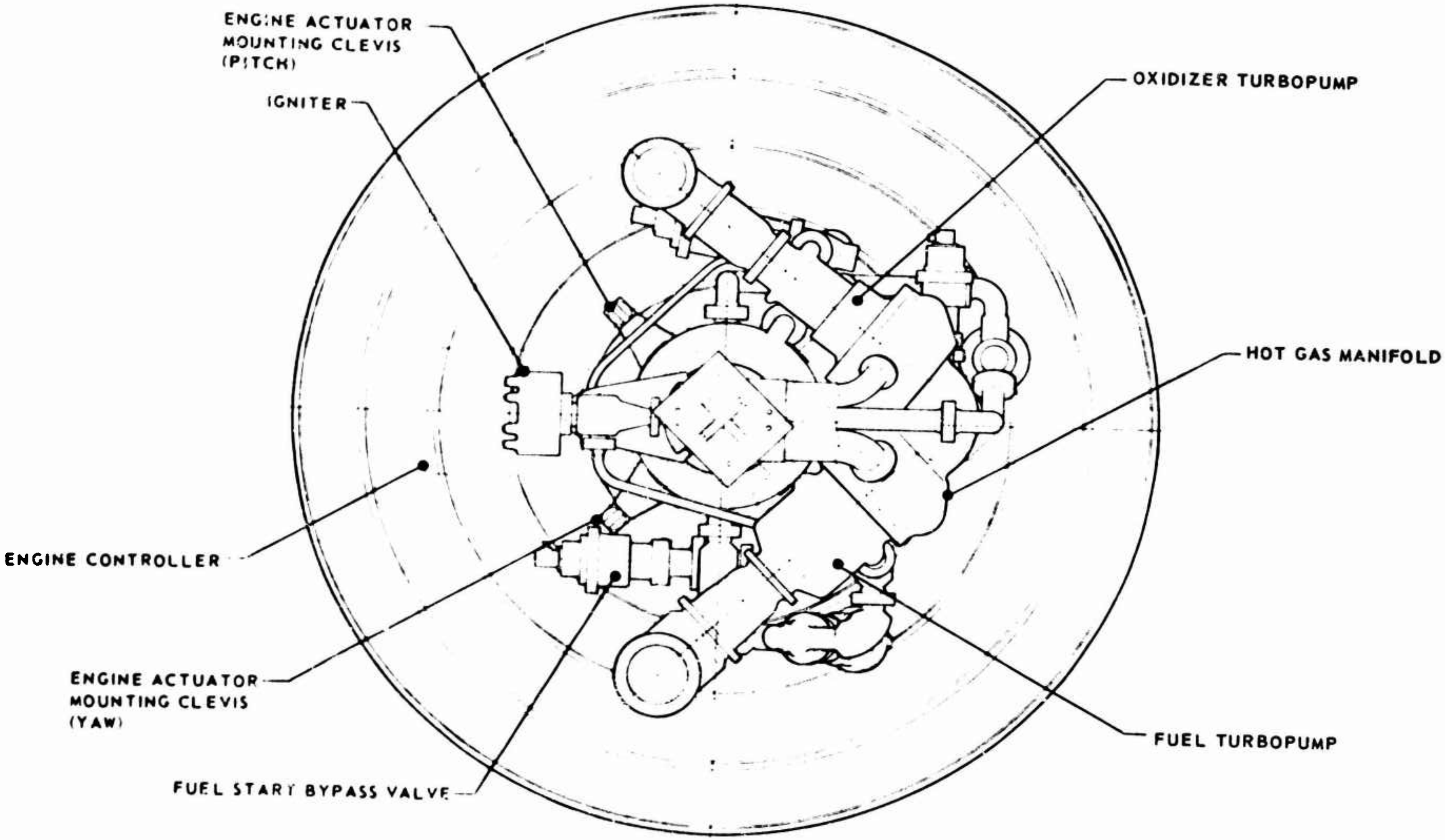


Figure 301. 10K Baseline Engine Configuration, Top View

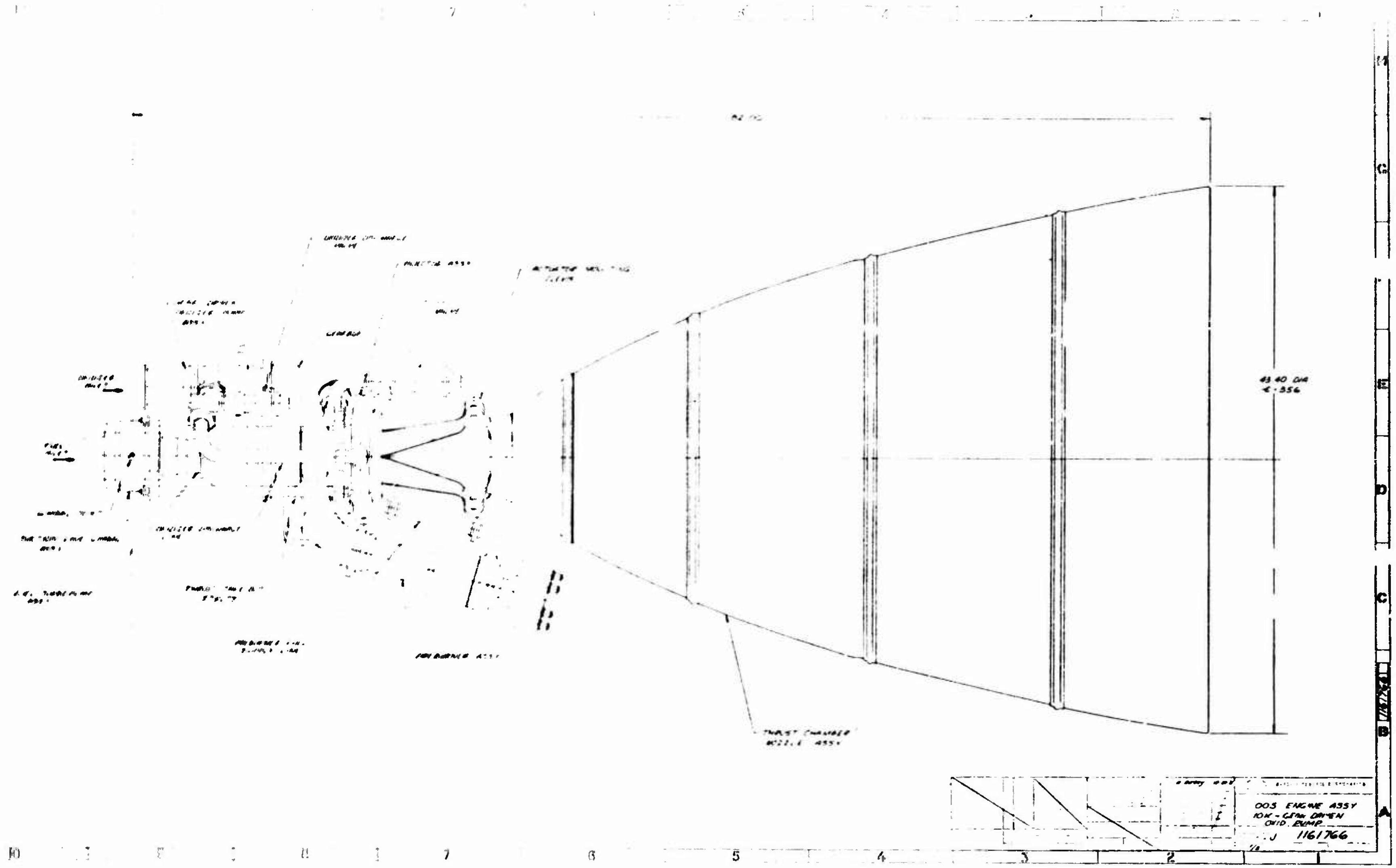


Figure 302. 10K Engine In-Line Fuel Pump (Sheet 1 of 4)

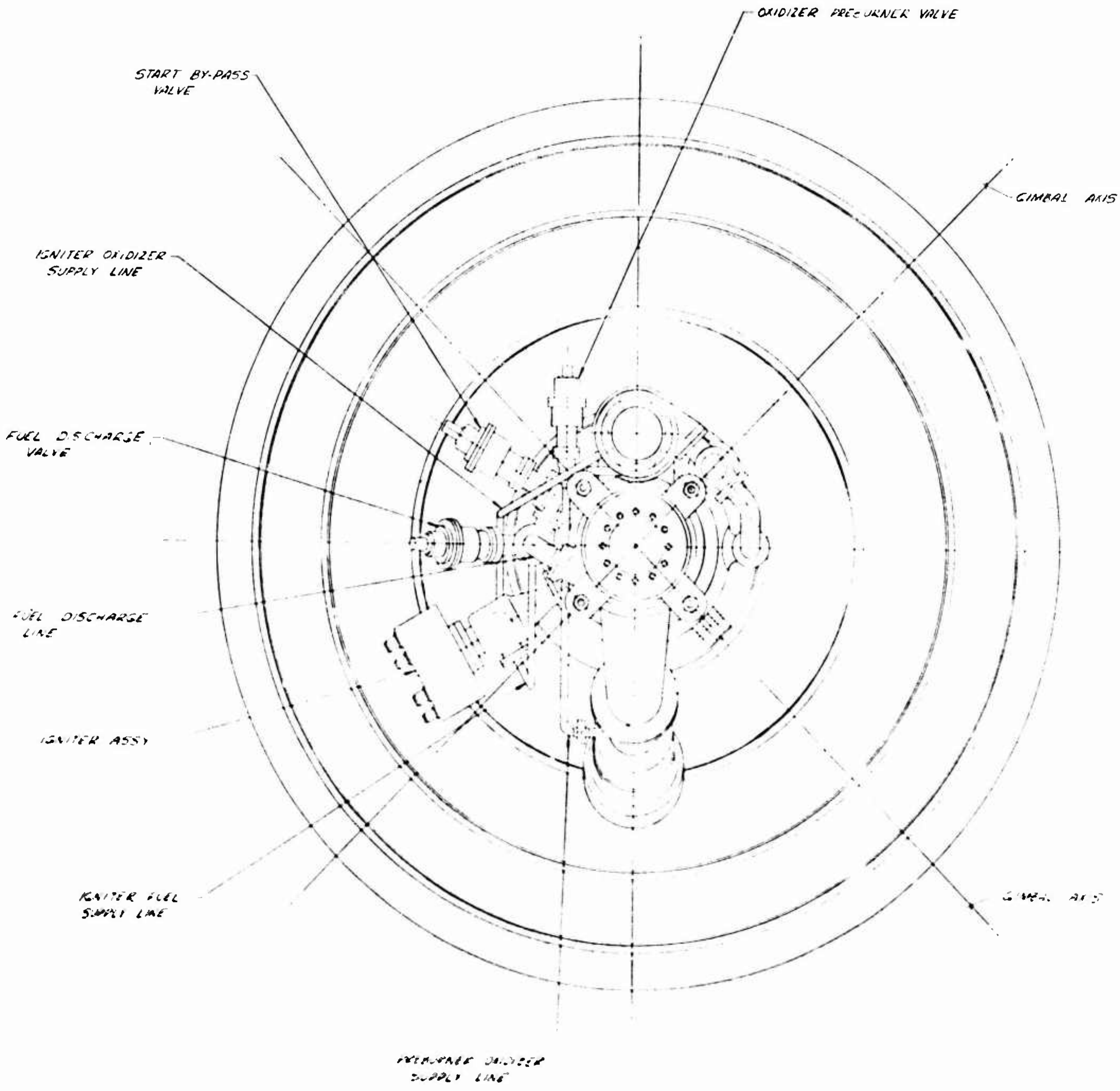


Figure 302. 10K Engine In-Line Fuel Pump (Sheet 2 of 4)

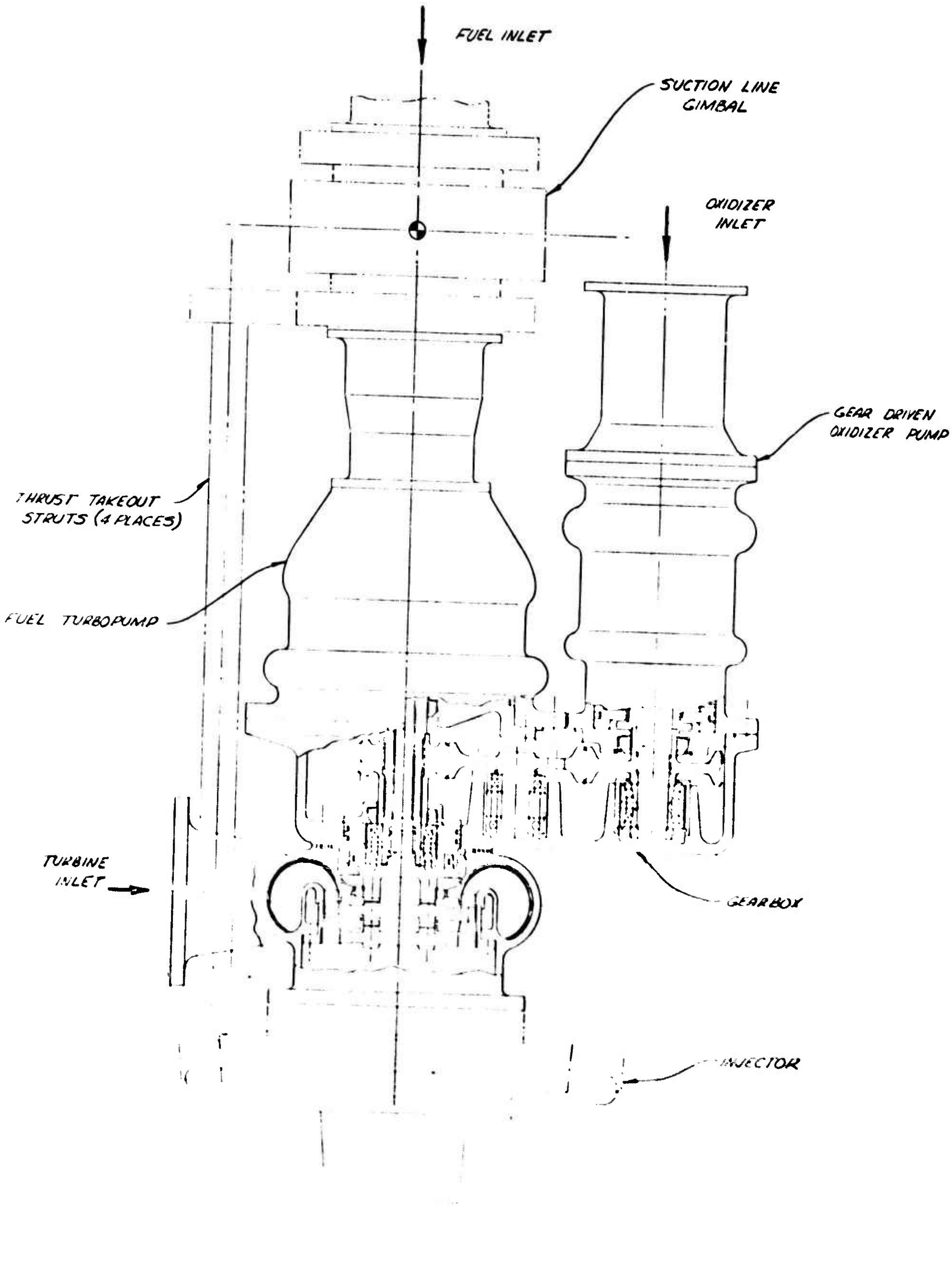


Figure 302. 10K Engine In-Line Fuel Pump (Sheet 3 of 4)

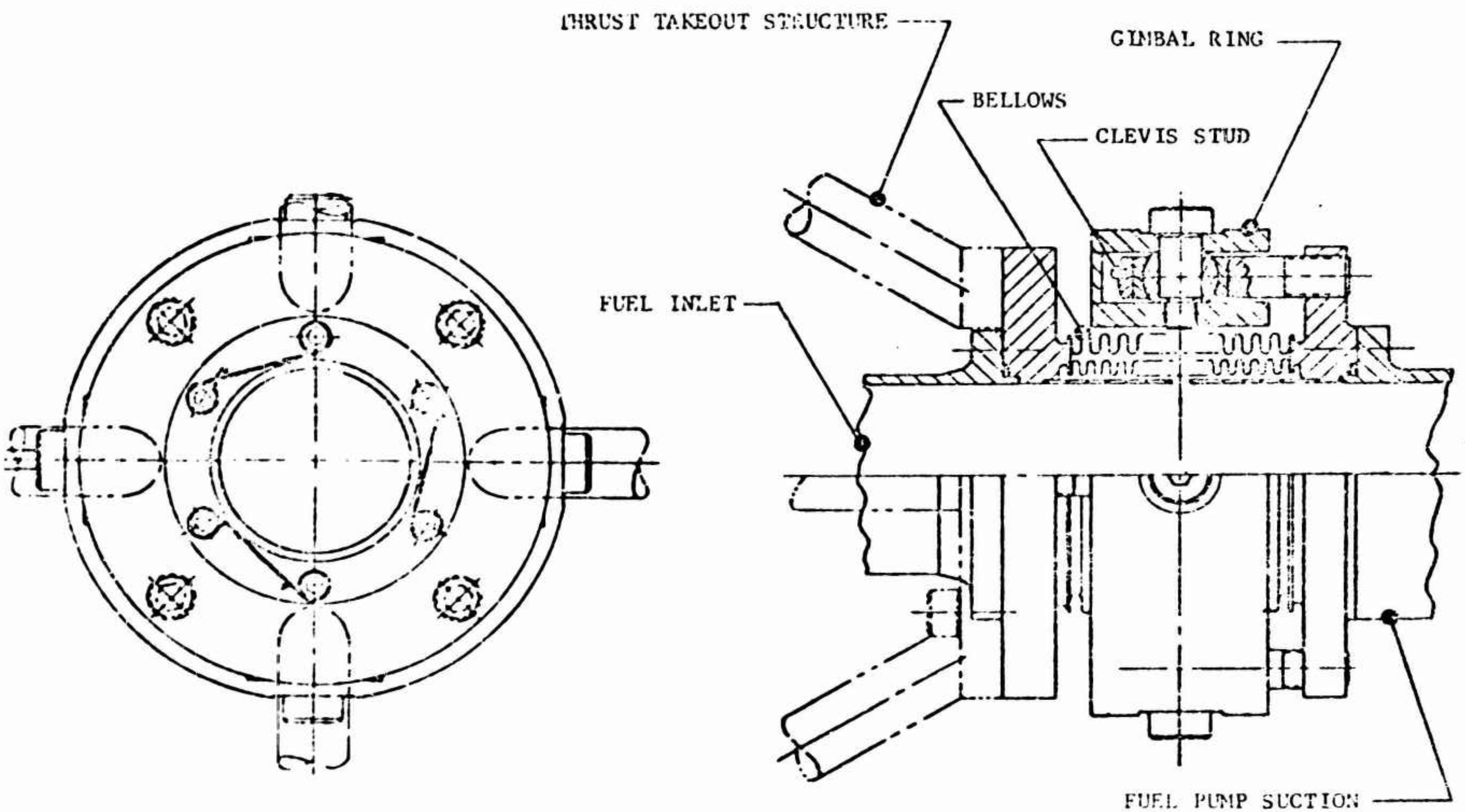


Figure 302. 10K Engine In-Line Fuel Pump (4 of 4)

TABLE CII

10K ENGINE DESIGN SUMMARY

Cycle: Staged Combustion

		Side Mounted Baseline	In Line Gear Drive
Performance			
F, (lb)		10,000	10,000
MR		6.0	6.0
I_s (sec)		465.5	465.0
P_c, (psia)		1250	1250
F/P_c		8.0	8.0
R_t (in.)		1.13	1.13
Envelope			
ϵ_o		400	356
ϵ_{TRANS} (Copper Chamber to Tubular)		6:1	6:1
Engine Overall Length (in.)		76	82
(L' = 5 in.) (L' Throat - Gimbal = 13.0 in.)			
Engine Stowed Length (in.)		76	82
Engine Exit dia (in.)		45.20	43.4
Engine Weight, W_{BO} (lb)		278.1	223
Suction Lines			
Diameter	LO_2/Fuel	2.41/2.77	2.41/2.77
Location Diameter	LO_2/Fuel	16.0/16.8	0/5.5

TABLE CIII

10K ENGINE WEIGHT SUMMARY BASELINE ENGINE

Side Mounted TPA

P_c = 1250 psia MR = 6.0 ε = 400:1 NPSH ft = 60(F); 16(O)

			Weight lb	Weight %
I.	Thrust Chamber			
	A.	Injector	12.4	
	B.	Copper TC to ε = 6:1	23.6	
	C.	Regen Tubes to ε = 400:1	72.2	
	D.	Ignitor	12.1	43.30
II.	TPA's			
	A.	Fuel incl. Boostpump	22.7	
	B.	Oxidizer incl. Boostpump	26.30	17.60
III.	Valves		27.2	9.78
IV.	Gas Lines, Gas Manifold, Liquid Lines		55.4	19.90
V.	Preburner		18.7	6.72
VI.	Gimbal Assembly and Support		7.5	2.70
	Total Calculated Weight		278.1 lb	100

Estimated Harness Instrument Support, Brackets and Attached Hardware	29 lb
Estimated Engine Controller	35 lb
Total Estimated Weight	342.1

TABLE CIII (cont.)

Gear Drive Concept

P_c = 1250 psia MR = 6.0 ε = 356:1 NPSH (H) = 60 F/160

		Weight lb	Weight % calculated
I.	Thrust Chamber		
	A. Injector	12.40	
	B. Copper Thrust Chamber	23.60	
	C. Regen Tubes to ε = 356	64.0	
	D. Igniter	12.1	50.1
II.	TPA's		
	A. Pumps, Boost Pumps and Gearbox	48.30	21.6
III.	Valves	27.20	12.20
IV.	Gas and Liquid Lines	9.70	4.30
V.	Preburner	18.70	8.40
VI.	Gimbal Assbl. and Support	7.50	3.4
	Total Calculated Weight, lb	223.5	100.0%
Estimated Harness, Instrument Support Brackets and Attach Hardware		29.0	
Estimated Engine Controller		35.0	
	Total Estimated Weight	287.5	

The hydraulic resistance of the system and stability considerations will limit chamber pressure to a value 30-50% of the minimum tank pressure. With saturated propellants, tank vapor pressures of 15 psia will result in thrust levels of 30-40 lb. It is desirable to minimize the need for feedback controls during idle mode and Table CIV shows a comparison of several operating points with fixed oxidizer valve positions. This shows that mixture ratio will remain in the satisfactory range of 1.5 - 3 without control of the oxidizer valve. Figures 303 and 304 show the effect of changing vapor pressures in the fuel and oxidizer tanks. This indicates that some control of oxidizer valve position would be required to compensate for large changes in tank pressure.

The thrust level shown in Table CIV is 0.3% of full thrust operation. The relative low level is due to the relatively small chamber throat size of high pressure engines.

Table CIV also shows the idle mode operation with externally pressurized oxygen and fuel tank pressure (Case 4). This case is of interest for systems which have stored tank pressurization capability on board which enables these systems to start as soon as the pumps are sufficiently chilled down.

A summary of the engine operating mode and control requirements for the complete engine start sequence is presented in Table CV for this engine start of sequenced pressure fed and pump assisted idle mode.

b. Engine Configuration Modification for Pump Assisted Idle Mode

The incorporation of the pump fed idle mode for autogenous tank repressurization requires that the LO_2 vaporizer be operative during idle mode. To meet this requirement, the LO_2 vaporizer is located around the thrust chamber since this is the only heat generating component during the idle mode operation. In Figure 305 the LO_2 vaporizer concept is shown. The LO_2 passages are machined into the closure wires of the fuel manifold and thus are avoiding any direct interpropellant leakage. The 2-dimensional heat transfer of the copper chamber will help to vaporize the oxidizer.

This modification will impact the engine weight only slightly and is estimated to be 4 lb. The engine life will not be impacted due to idle mode operation. Section III.E.8 presents the cost impact due to the idle mode requirement. A heat transfer analysis was conducted to establish the surface requirements of the LO_2 vaporizer and tank pressurant condition during idle mode. The results are summarized in Table CVI.

TABLE CIV

COMPARISON OF OPERATING POINTS WITH FIXED OXIDIZER VALVE POSITIONS

1) Start of childown
2) End of chilldown, jacket and injectors at 520°R
3) End of chilldown, jacket and injectors thermal equilibrium
4) Same as (3) but externally pressurized tanks

CASE	1	2	3	4
THRUST, LB.	35.0	34.0	33.1	42.4
MIXTURE RATIO	1.7	1.9	2.6	3.0
SPECIFIC IMPULSE, SEC.	407	412	431	442
FUEL TANK PRESSURE, PSIA	15	15	15	17
NET POSITIVE SUCTION HEAD, FT.	-	0	0	60
ΔP COOLING JACKET AND BYPASS, PSI	9.1	9.3	9.5	10.4
ΔP TURBINE AND BYPASS	0.1	0.2	0.2	0.2
ΔP GAS INJECTOR, PSI	0.6	0.5	0.6	0.5
TOTAL FUEL FLOW, LB/SEC	.031	.029	.021	.024
COOLING JACKET FLOW, LB/SEC	.016	.029	.021	.024
COOLING JACKET BYPASS FLOW, LB/SEC	.015	0	0	0
PREBURNER BYPASS FLOW, LB/SEC	.011	.018	.015	.017
PREBURNER FUEL FLOW, LB/SEC	.005	.010	.006	.007
OXID. TANK PRESSURE, PSIA	15	15	15	23
NET POSITIVE SUCTION HEAD, FT.	0	0	0	16.0
ΔP OX. VALVE, PSI	8.6	8.7	9.3	15.7
ΔP OX. TC INJECTOR, PSI	1.2	1.3	1.0	1.4
CHAMBER PRESSURE, PSIA	5.2	5.0	4.7	5.9
OX TC VALVE ADMITTANCE, (KW)	.0172	.0172	.0172	.0172
COOLING JACKET BYPASS ADMITTANCE (KW)	.80	0	0	0
PREBURNER BYPASS ADMITTANCE (KW)	3.0	3.0	3.0	3.0

Pressure Fed Idle Mode

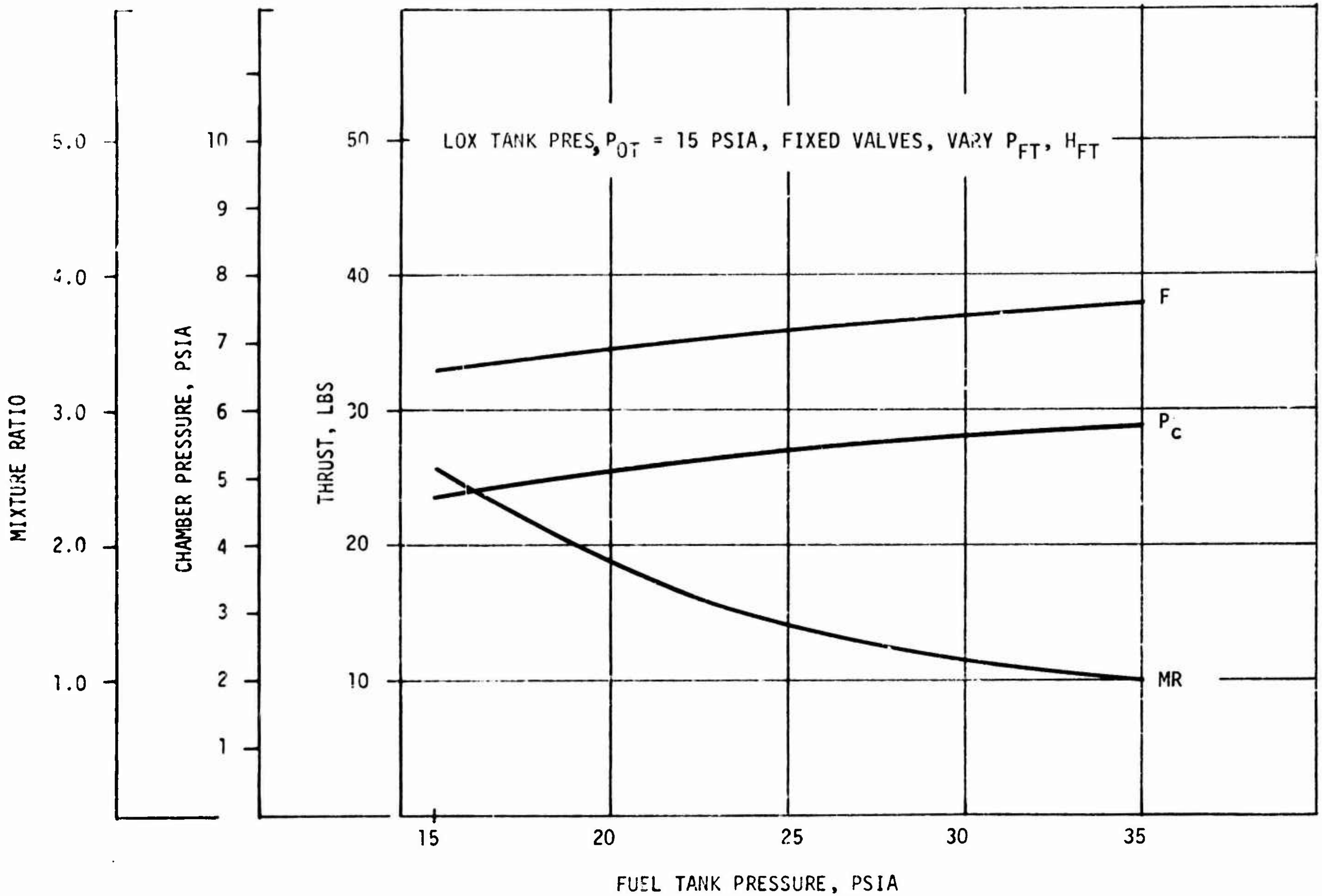


Figure 303. Pressure-Fed, Idle Mode, Zero NPSH-Fuel

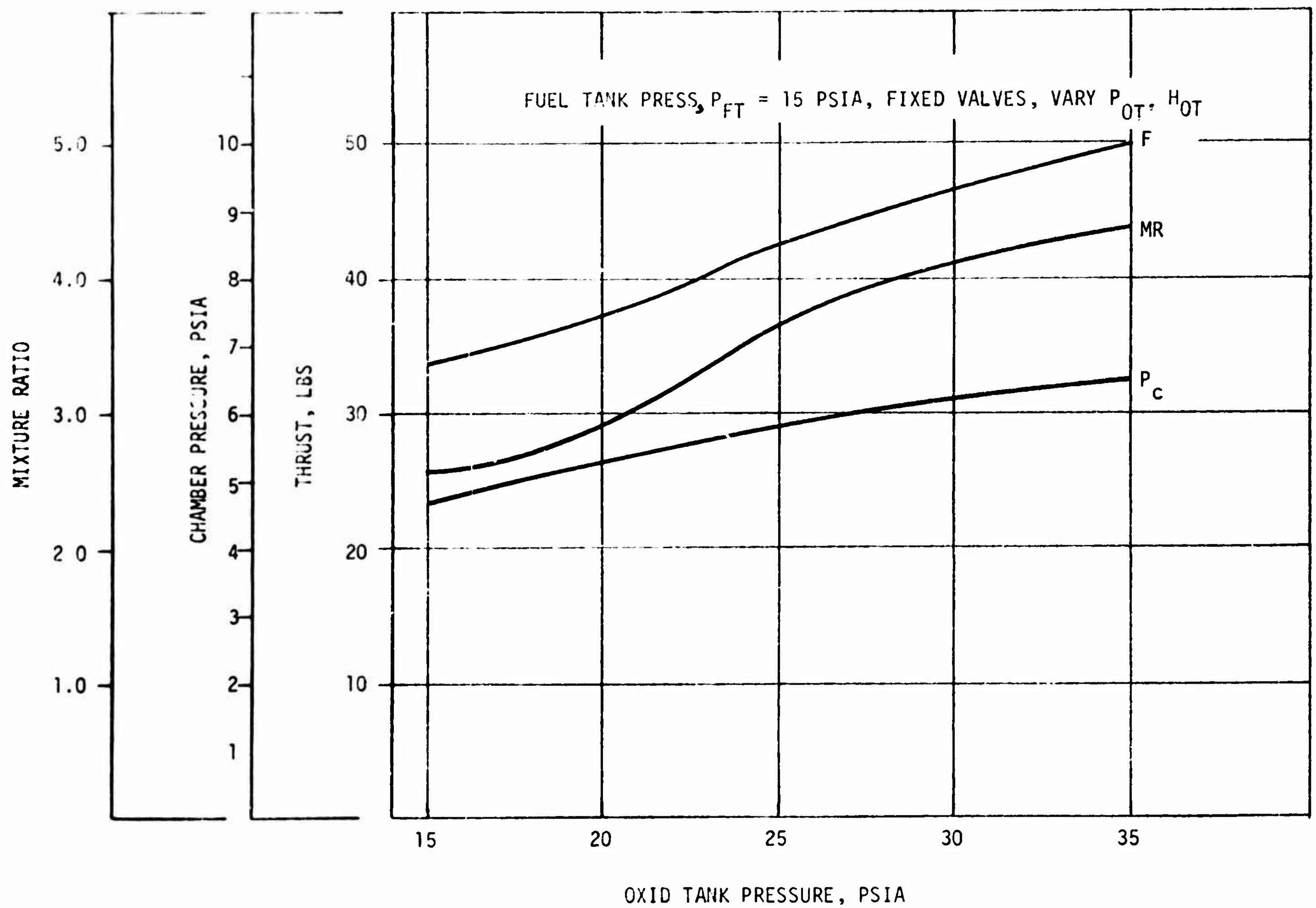


Figure 304. Pressure-Fed, Idle Mode, Zero NPSH-Oxidizer

TABLE CV

IDLE MODE OPERATION 10K ENGINE DESIGN

OPERATING MODE	TANK CONDITION	ENGINE OPERATING SEQUENCE	ENGINE CONTROL	CONSTRAINTS
Pressure fed idle mode	∘Zero NPSH Variable LOX Press Variable Fuel Pressure ∘Settled Propellant	∘H_2 Main Valve Full Open ∘Main Chamber Igniter On ∘LOX Main Valve Idle Setting	∘Fixed LOX Main Valve pos. depending on tank pressures. ∘Preburner Bypass Open ∘Jacket Bypass Modulating closed ∘LOX GG Valve Closed	Thrust Level depending on fuel tank pressure
Pump Assisted Idle Mode	Initiation: Zero NPSH LOX Tank Pressure increasing = MR shift and thrust increase Fuel Tank Pressure increase depending on LOX Tank Press.	∘Close GG Bypass Valve. ∘Open LOX Autogenous Valve ∘Open Fuel Autogenous	∘Modulate GG Bypass closed ∘LOX Main Valve Set for Pump assist idle	Initiation at Zero NPSH at LOX and Fuel Pump Suction TPA Breakaway Torque Fuel TPA Speed Limit LOX TPA Speed Limit TCA Mixture Ratio Main Injector 2 Phase Flow
Engine Start Initiation	Start NPSH	∘Light Preburner ∘Min. GG MR Setting	∘Modulate GG Bypass Closed ∘Open GG LOX Valve ∘Open LOX Main Valve	Initiate at reaching Tank Pressure Start Conditions

TABLE CV (cont.)

PUMP FED IDLE MODE   0 (NPSH)   F = 30.9 LB

O_2: P_{OT} = 15 PSIA
P_{IN} = 15.01 PSIA
T_{IN} = 162.6 °R
W_o = 0.0454 LB/SEC

H_2: P_{FT} = 18.057 PSIA
P_{IN} = 18 PSIA
T_{IN} = 37.5 °R
h = -106.2
w_f = 0.035 LB/SEC

ENGINE: MR = 1.291
P_c = 4.65 PSIA

TANK PRESSURE IDLE MODE

F = 47.75 LB   MR = 6.91 P_c = 6.9 PSIA

FUEL	OXID	AUTOGENOUS PRESSURE FLOW
N_T = 6683 RPM	W_o = 0.4297 LB/SEC	W_o = 0.352 LB/SEC
P_D = 33 PSIA	P_D = 26.75 PSIA	
T_D = 37.45 °R	T_D = 162°R	N_o = 91 (NO. CHANNELS)
W_f = 0.0614 LB/SEC	N_T = 3179 RPM	A_o = 0.001 in.^{2} PER CHANNEL

P_D = PUMP DISCHARGE PRESSURE
T_D = PUMP DISCHARGE TEMP

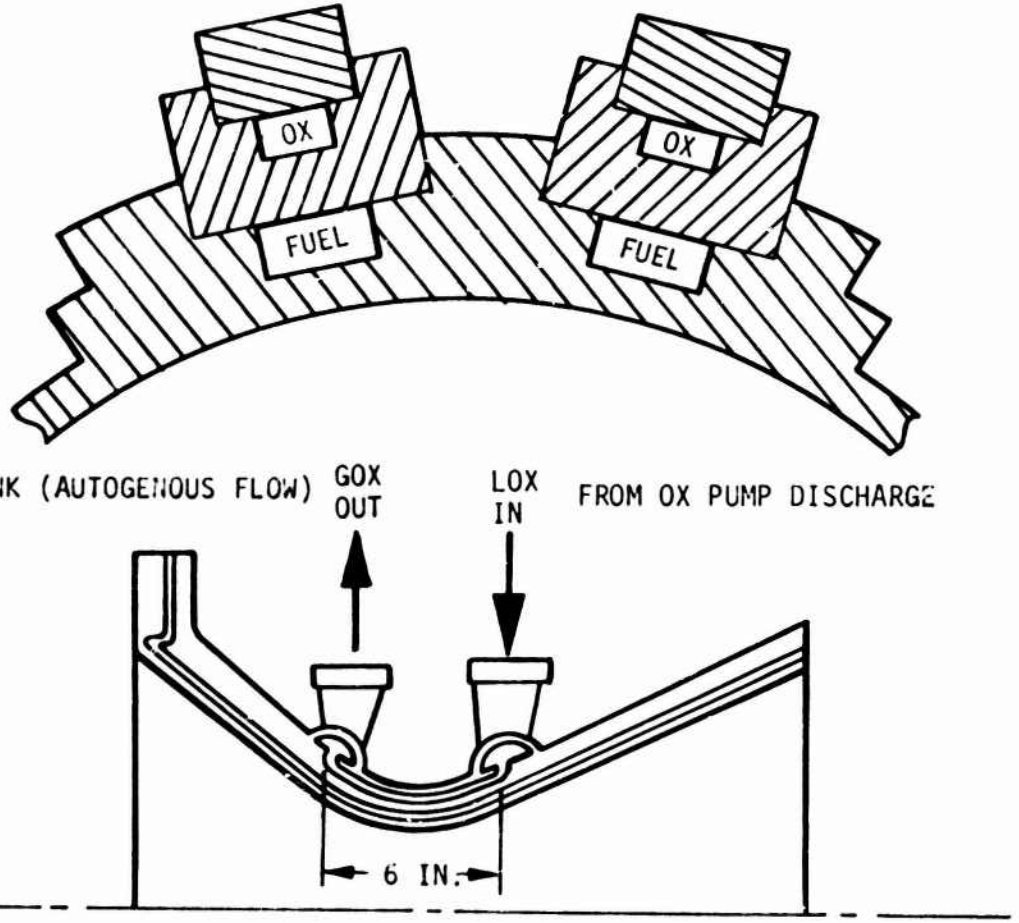


Figure 305. LOX Vaporizer Concept for Autogenous Pressurization System

TABLE CVI

PUMP FED IDLE MODE OXIDIZER AUTOGENOUS SYSTEM HEAT TRANSFER SUMMARY

Thrust Chamber	
Chamber Pressure	7.5 psia
Mixture Ratio	2.1
Total Chamber Flow	0.114 lb/sec
Cooling Jacket Flow (LH_2)	0.0368 lb/sec
Coolant Inlet Pressure	33 psia
Coolant Inlet Temperature	42°R
Predicted Coolant Temperature Rise	450°R
Predicted Coolant Pressure Drop	15 psi
Tank Pressure, Fuel	17 psia
Tank Pressure, LO_2	23 psia
Autogenous System	
Number of Channels	91
Channel Flow Area	0.00071 $in.^2$
Oxidizer Flow	0.35 lb/sec
Oxidizer Inlet Pressure	27 psia
Oxidizer Inlet Temperature	170°R
Predicted Total Enthalpy Rise	80 Btu/lb
Predicted Pressure Drop	12 psi
Heat Exchanger Length	6 in.

c. Pump Fed Idle Mode Operation

An investigation was conducted of the feasibility of a pump assisted idle mode for the 10K engine. This does not effect chilldown time, since the engine cannot be started until the pumps have been chilled and saturated liquid is available at pump suction, however, it provides a means of autogenous tank pressurization to bring NPSH up to a value permitting a normal engine start without any external source of tank pressurization.

The sequence of events for initiating the pump fed idle mode is as follows:

1. Engine is operating as in Case 3 of Table XCIX at end of chilldown.

2 Preburner fuel bypass valve is closed, sending all fuel flow through the turbines and developing turbine torques of 2.7 and 2.0 in.-lb for the fuel and oxidizer turbines.

3. After pump rotation has begun, the autogenous valves are opened permitting tank pressurization.

4. Once the tank pressures reach the required levels, the normal engine start sequence begins. If, for any reason, a "hold" is required prior to engine start, the preburner bypass valve may be partially reopened to obtain a steady state thrust in the 50-100 lb range.

Operation of a pump fed idle mode at zero NPSH depends on the existance of thermodynamic head which is a function of propellant enthalpy. Table CVII shows operating points at start and completion of tank pressurization. These are steady state points with power balanced by the autogenous flow or preburner bypass. Shutting off the autogenous flow partially unloads the pumps and increases turbine speed. Limiting speeds for the fuel and oxidizer pumps at zero NPSH are approximately 12,000 and 4000 rpm. When operated in the engine, higher speeds are possible at higher flow coefficients but the flow must be recirculated; if it is injected in the thrust chamber, the increase in pressure will force the pumps into operation at low flow coefficients with poor cavitation performance.

It should be noted that the two cases in Table CVII are at the beginning and completion of autogenous pressurization. Because the fuel autogenous tap-off is at the cooling jacket exit, Case 1 shows less than 1 psi driving pressure for fuel pressurant flow. This is satisfactory

TABLE CVII

PUMP ASSISTED IDLE MODE

CASE NO.	TANK PRESSURE = VAPOR PRESSURE 1		FULLY PRESSURIZED PROP. TANKS 2	
	FUEL	OXID.	fuel	OXID
TANK PRESSURE, PSIA	15	15	16.8	22.8
TURBINE SPEED, RPM	6680	3180	9060	5480
HIGH SPEED INDUCER				
NPSH FT (NO TSH)	0	0	60	16
THERMODYNAMIC SUCTION HEAD	113	4	113	4
SUCTION SPECIFIC SPEED	470	1700	544	512
FLOW COEFFICIENT	.30	.46	.31	.08
FRACTION CAVITATION LOSS	0	0	0	.0002
MAIN STAGE PUMP				
NPSH, FT (NO TSH)	15	4	96	29
THERMODYNAMIC SUCTION HEAD	115	4	115	4
SUCTION SPECIFIC SPEED	475	1190	560	540
FLOW COEFFICIENT	.33	.51	.34	.16
FRACTION CAVITATION LOSS	0	0	0	0
FINAL STAGE PUMP DISCH. PRESS.	33.1	26.6	50.6	60
AUTOGENOUS SUPPLY PRESSURE	15.8	26.6	28.7	60
AUTOGENOUS FLOW RATE, LB/SEC	.02	.35	0	0
CHAMBER FLOW RATE, LB/SEC	.037	.078	.087	.123
COOLING JACKET FLOW RATE	.037	-	.087	-
PREBURNER FLOW RATE	.037	0	.069	0
PREBURNER BYPASS FLOW RATE	0	-	.019	-
GAS INJECTOR EXIT TEMPERATURES, °R	424	481	380	363
PREBURNER PRESSURE	10.8		19.7	
TURBINE EXHAUST PRESSURE	7.5		14.0	
THRUST	47.8		83.2	
MIXTURE RATIO	2.1		1.4	
SPECIFIC IMPULSE	419		396	
[illegible]. SEC VALVE KW	.0172		.0172	
PREBURNER BYPASS VALVE KW	0		.300	

since the limiting pump is the oxidizer pump. As the oxidizer tank is pressurized, both pump speeds and chamber pressure will increase thus providing adequate fuel tank pressurant flow. If Case 2 had the fuel tank pressure at 15 psia, the fuel pump would still be operating in a non-cavitating condition. The actual transient was not run on the computer because of time and budget limits but it appears that no control operation other than the autogenous valves (and possibly the preburner bypass valve) would be required.

6. Effect of Engine Cycle Life on Engine Design Point

The engine low cycle fatigue cycle requirements have a considerable effect on the engine design point selection. The engine components most effected by these requirements are the thrust chamber and fuel turbine disks.

The turbine disk life can be manipulated by the selection of turbine inlet temperature and turbine tip speed identical to the relationship presented for the 25K thrust engine. The variations of these two turbine design parameters effects the power balance and therefore chamber pressure capability.

The chamber life requirement is most effectively manipulated by changing thrust chamber pressure and coolant throat Mach. No. The parametric analysis yielded the design information required to establish the coolant pressure drops for each chamber pressure required to meet the 60 and 600 life cycle requirement. The data is summarized in Figure 306.

With this pressure drop requirement and the selected turbine temperatures and tip speed, the power balance was conducted over the appropriate chamber pressure range. The chamber pressures were then selected meeting the chamber life requirements and power balance capability. The flow, temperature and pressure schedules are shown in Figure 307 and Figure 308 and the turbopump operating conditions are summarized in Table CVIII.

The chamber pressures of P_c = 1050 psia for 600 cycles and P_c = 1400 psia for 60 cycles were defined. This change in chamber pressure effects engine performance weight and envelope. Utilizing the parametric engine data, the engine characteristics can be summarized as follows:

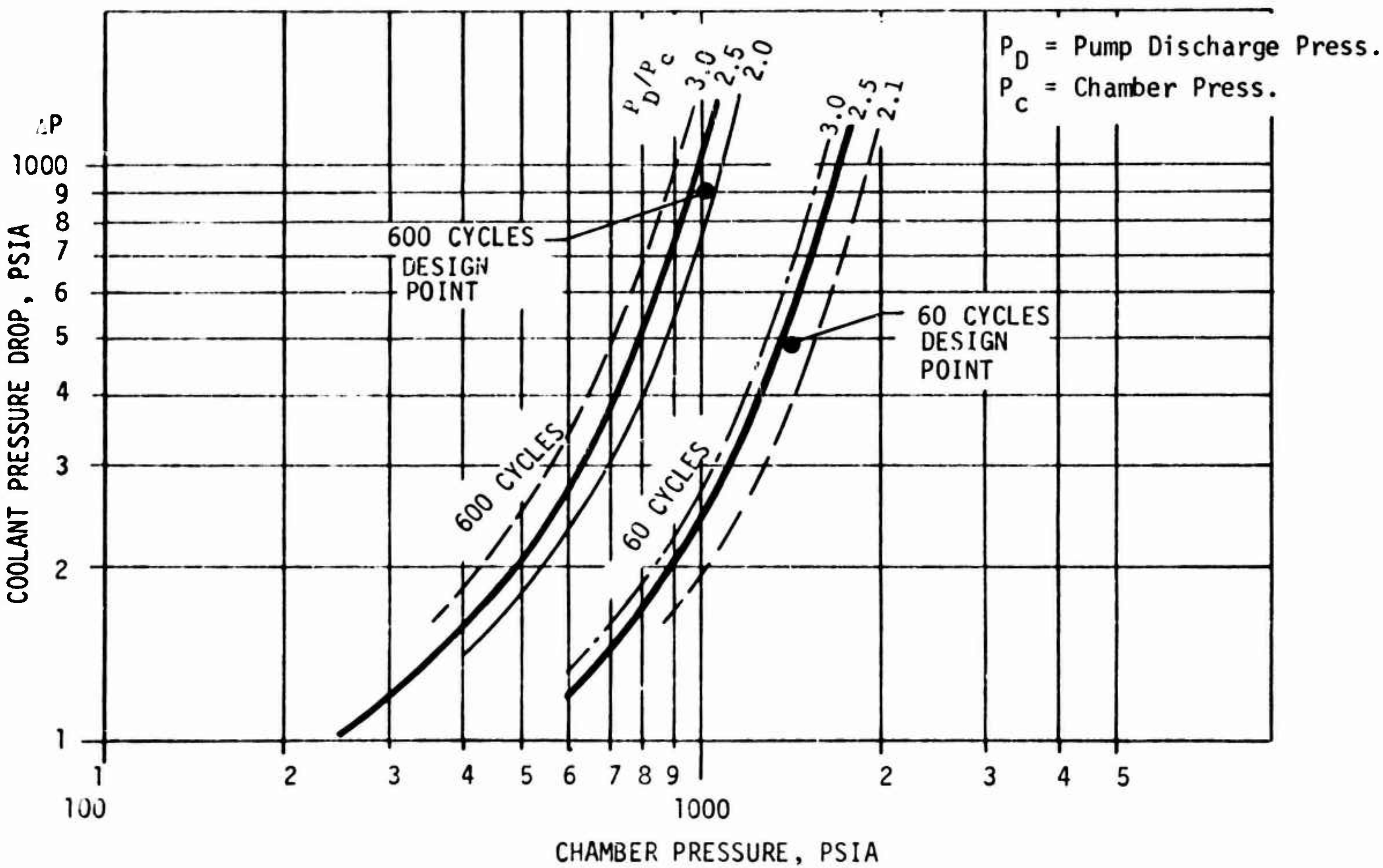


Figure 306. 10K Engine, Effect of Chamber Cycle Life on Coolant Pressure Drop

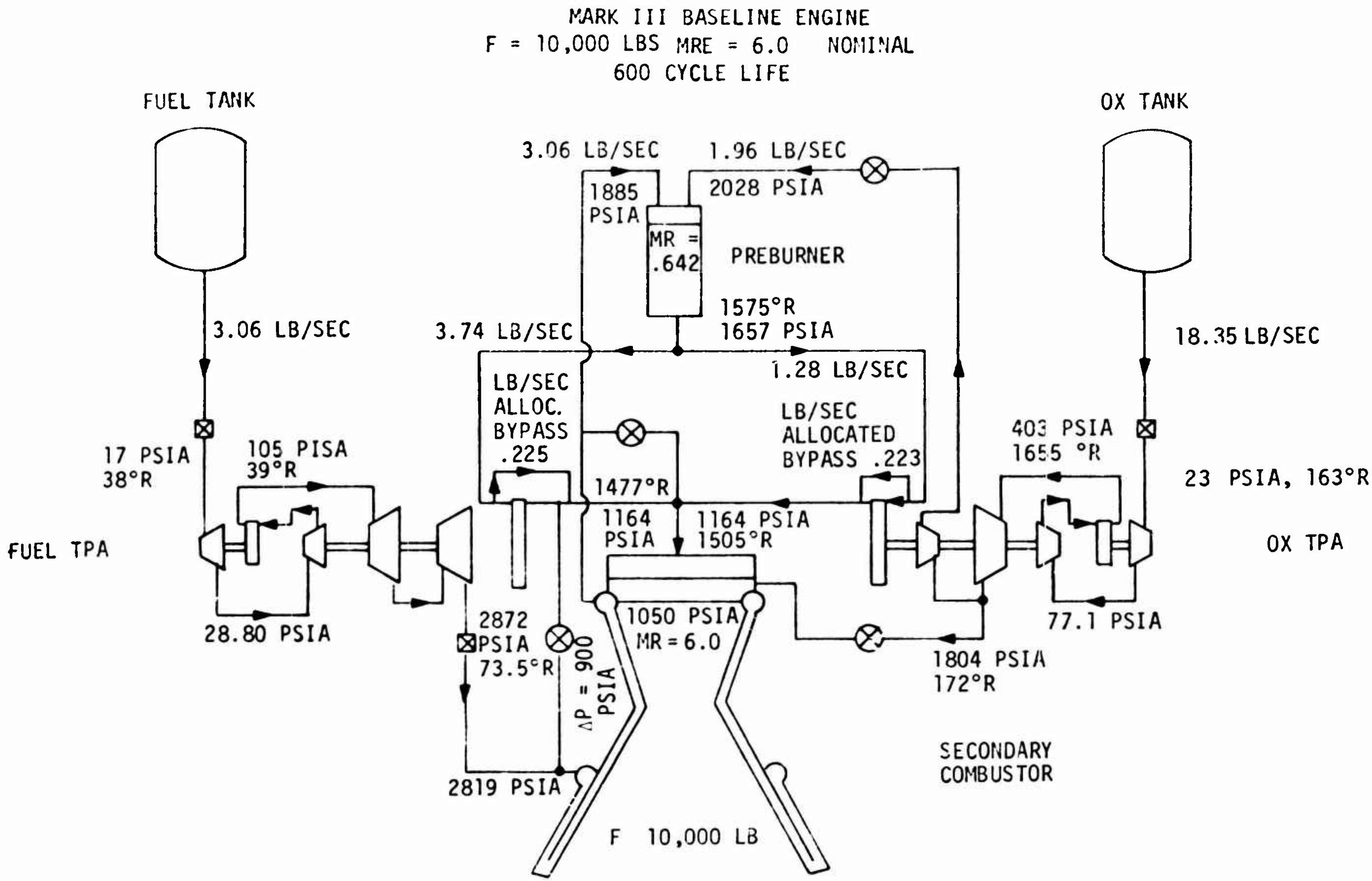


Figure 307. 10K Engine Flow Temperature and Pressure Schedule for 600 Cycle Life

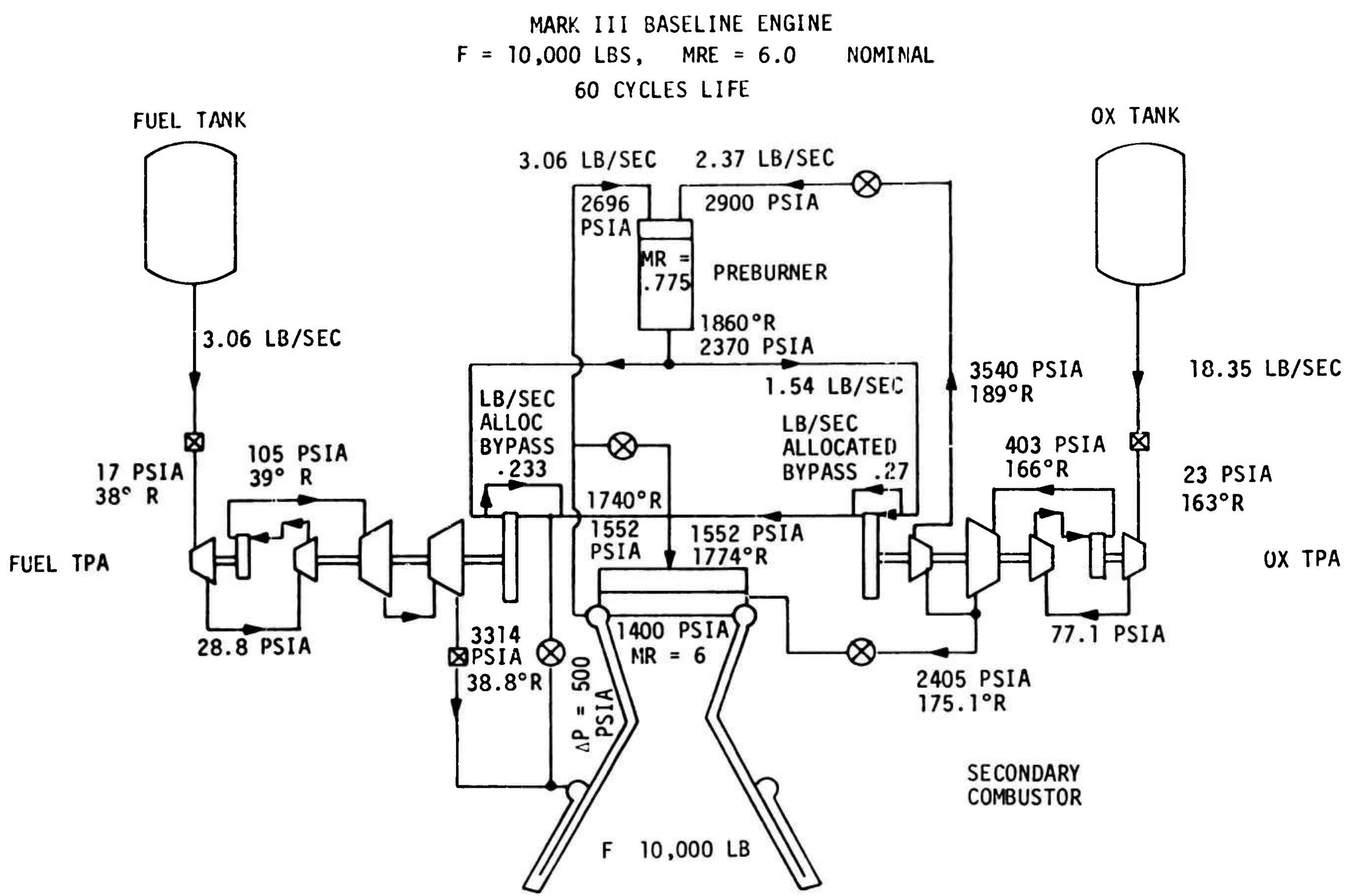


Figure 308. 10K Engine Flow Temperature and Pressure Schedule for 60 Cycle Life

TABLE CVIII

OOS 10K ENGINE CYCLE-LIFE SENSITIVITIES

	60 Thermal Cycles		300* Thermal Cycles		100,000 rpm Fuel TPA 600 Thermal Cycles		87,500 rpm Fuel TPA 600 Thermal Cycles	
	Oxid	Fuel	Oxid	Fuel	Oxid	Fuel	Oxid	Fuel
Turbine Inlet Temp, °R	1860	1860	1860	1860	1575	1575	1575	1575
Turbine Mean Blade Speed, ft/sec	1100	1300	1100	1300	1100	1650	1100	1650
Thrust Chamber Pressure, psia	1400		1250		1050		1050	
Regen. ΔP, psi	500		859		900		900	
Regen. ΔT, °R	465		447		423		423	
Pump Discharge Pressure, psia	2405**	3314	2148**	3281	1804**	2872	1804**	2900
Delta TPA Weight, Pound	+0.9906Δ	+0.3738Δ	Ref.	Ref.	-1.2901Δ	+1.2598Δ	-1.2417Δ	+9.6888Δ
Delta Boost Pump Weight, Pound	0 Δ	0. Δ	Ref.	Ref.	0. Δ	0. Δ	0. Δ	+0.0551Δ
Delta TPA and Boost Weight, Pound	+0.9906Δ	+0.3738Δ	Ref.	Ref.	-1.2901Δ	+1.2598Δ	-1.2417Δ	+9.7439Δ
Suction Line Diameter, in.	2.41	2.77	2.41	2.77	2.41	2.77	2.41	2.90
Delta TPA and Boost Length, in.	+0.0175Δ	+0.0023Δ	Ref.	Ref.	-0.0267Δ	+0.5198Δ	-0.0267Δ	+1.7927Δ
Main TPA Shaft Speed, rpm	63,000	100,000	63,000	100,000	63.000	100,000	63,000	87,500
Thrust Bearing Maximum Axial Force (Includes Preload) lb	74+	60+	48	38	32	6.5	32	34

* Reference Run 9 Oct 71 18:15:24 and 11 Oct 71 15:00:52 Baseline Turbopump

**First-stage Pump

Engine Characteristics for 60 and 600 Cycle Life

	60 Cycles	600 Cycles
Thrust, lb	10K	10K
Chamber Pressure, psia	1400	1050
Area Ratio	400	400
Engine Length, in.	73	81
Engine Diameter, in.	43.0	49.60
Specific Impulse, sec	466.20	464.8
Engine Weight Change, lb	-10	+16
TPA Weight Change, lb	0.0	+1.7
F/P_c	7.14	9.53
Total Engine Weight, lb (W/O Harness & Engine Controller)	268	294

The engine payload change due to changing life requirement is

$$\Delta PL = \frac{\partial PL}{\partial I_s} \times \Delta I_s + \frac{\partial PL}{\partial W_t} \Delta W_t$$

	60 Cycles	600 Cycles
Payload Change lb	+184	-243.0

7. Elimination of Throttling Requirement

The elimination of the throttling requirement can result in minor engine configuration modifications. The preburner bypass valve and line will be eliminated resulting in a weight savings. The preburner control valve resistance is reduced to 200 psia and the preburner LO_2 circuit pressure drop is increased to 800 psia and liquid LO_2 injection will be used. The LO_2 vaporizer for the preburner is eliminated and replaced by a hydrogen regenerative section.

To maintain autogeneous capability, a LOX vaporizer has to be available and is recommended to be placed around TCA, which could maintain the capability to operate at pump assisted idle mode.

A modified engine schematic is shown in Figure 309. The weight change due to elimination of the throttling requirement is $-\Delta W = -8.20$ lb.

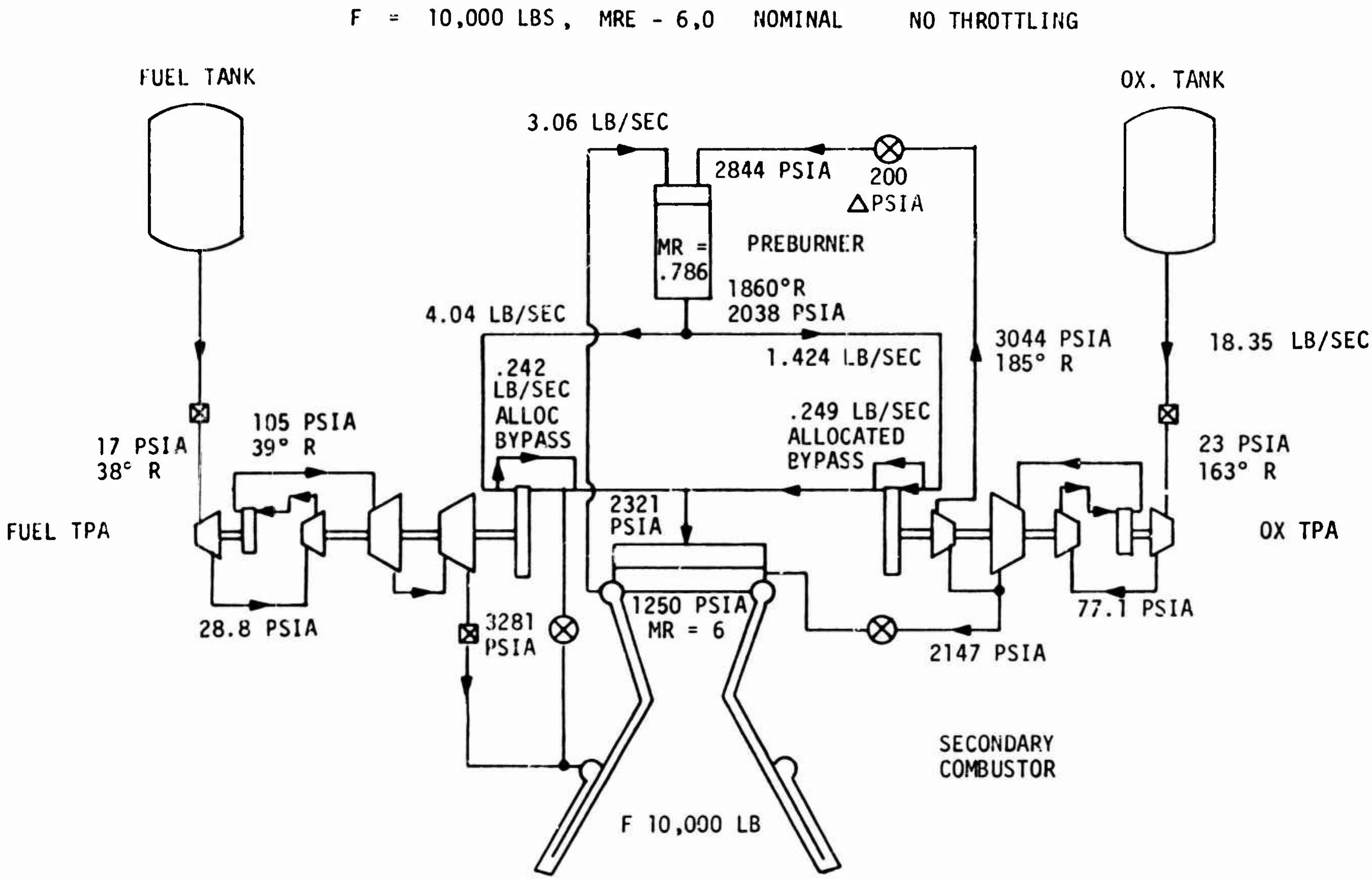


Figure 309. 10K Engine Temperature, Pressure, and Flow Schedule-No Throttling

8. Engine Development and Cost

a. Baseline Engine

The development of the 10K engine will be accomplished within the basic program schedule shown in Figure 310 for the 25K engine and it is anticipated that approximately the same amount of hardware and facility modifications will be required.

The program controls will also be identical to these used on the basic 25K engine effort, consequently, the costs for the basic 10K engine programs are expected to be identical to those of the 8K engine program. These are:

Demonstrator Program	$16,374,000
Development Program	38,993,000
Production (40 Units)	22,273,000
Total	$77,640,000
First Production Unit Cost	$ 708,000

Propellant requirements will be:

	Demonstrator Program	Development Program
LH_2	354,000 lb	1,890,000 lb
LO_2	1,110,000 lb	565,000 lb
LN_2	212 tons	1,130 tons
H_e	384 KSCF	2,050 KSCF

b. Effects of Varying Design Conditions

(1) Cyclic Life

The technical aspects of increasing cyclic life to 600 cycles/20 hr firing life or reducing to 60 cycles and 2 hr is discussed in Section III.F.6. That section shows the engine configurations to be very similar for either of the two conditions, consequently, the only variation to program costs will be associated with the increases or decreases to testing.

All of these costs will be incurred in the development portion of the program.

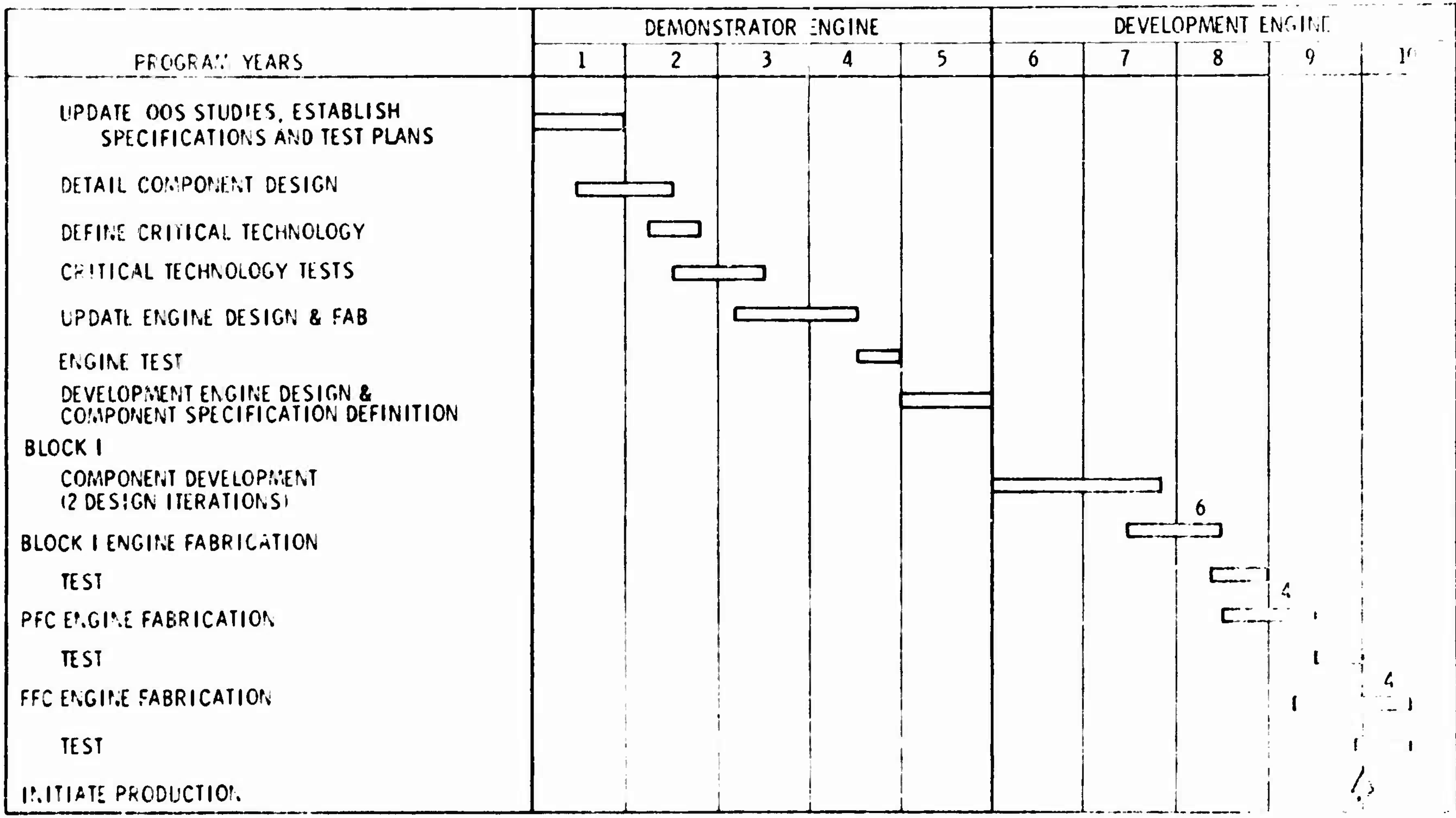


Figure 310. OOS Master Schedule

The increase in costs associated with the increase in cyclic life are estimated to be:

Hardware	$100,000
Testing	230,000
Total	$330,000

The reduction in costs with reduced cyclic life which would be applied to the development program are $230,000, all of which are a reduction in test costs.

(2) Idle Mode Operation

The only cost impact resulting from the idle mode new design condition is associated with the pumped case.

The additional costs are incurred in the development portion of the program due to the necessity of additional effort in testing the combustion chamber heat exchanger. These costs will be

Hardware	$210,000
Testing	460,000
Total	$670,000

(3) No Throttle Capability

The cost impact of eliminating the throttling capability as discussed in Section III.F.7 will be felt in two areas, development and engine production.

Development program costs will be reduced as follows:

Hardware	$308,000
Testing	690,000
Total	$998,000

Production unit engine costs will be reduced by approximately $20,000.

9. Major Component Design Description

The combustion components of the 10K thrust OOS engine are essentially scaled-down versions of the larger, 25K engine studied in the previous section. The operational requirements of the two engines are very similar, except for the maximum thrust level, and the depth of examination required in certain areas of extended requirements.

a. Engine Scaling Considerations

Most engine component weights and envelopes do not scale down directly with maximum delivered thrust levels, even though the delivered specific impulse remains relatively constant with thrust scaling, so that propellant flow rates do scale linearly. When smaller components are designed to operate with the same thermal characteristics, e.g., wall and fluid temperatures are substantially the same as larger counterparts, as is the case with the two subject engines. The sizes and weights are largely dependent upon internal fluid pressures. Fluid pressure contributes greatly to the total stress level of many engine components, and, hence, to their wall thicknesses and weights. Whereever the mixture fluid is compressible, the fluid density and required flow area is also affected by pressure. The weight and volume of most combustion component parts are greatly affected by propellant flow area requirements, because they all handle propellants and require propellant cooling.

Every combustion component is a heat exchanger, using this fact as basis for selecting the proper scaling requires that local heat fluxes be similar. Given the same temperature data, it is then required to match fluid velocities (Mach numbers) and passage sizes (Reynolds numbers). Power balance and thrust chamber life considerations require that the 10K engine operate at a lower chamber pressure than the 25K engine, because it is not possible to obtain similar Reynolds numbers within the chamber itself, otherwise. The use of lower system pressures in the 10K engine causes gas passage total flow areas to be designed greater than 10/25 = 40% as large as the 25K engine. Required flow areas are more nearly: $\frac{1800}{1250} \times \frac{10}{25} = 58\%$ of those used in the 25K engine. Those components whose wall thicknesses are are dependent upon pressure induced structural requirements will have walls nominally $(0.58)^{1/2} \times \frac{1250}{1800} = 53\%$ as heavy as the larger engine counterparts. Many component lengths are unchanged for reasons of heat transfer and/or combustion length requirements. Some, however, have envelopes and weights which are rather insensitive to any scaling parameters.

One such example is the main chamber igniter. In other cases, minimum fabrication gage determines material thicknesses, rather than structural or flow considerations.

Tables CIX through CXIV list the basic design specifications of the 10K engine combustion components. These components are shown in Figures 311 through 314. Preceeding the design specification (which includes calculated component weights) and the corresponding figure, is a brief discussion of the differences between the 10 and 25K component designs.

Component low cycle fatigue lives are very similar to those of their larger counterparts, because by design their thermal characteristics have been unaltered.

b. Main Injector

The 10K engine main injector is a propellant conditioning device as well as a metering and delivery component, as is the 25K unit. In order to duplicate the heat transfer properties of the larger injector, the 10K injector utilizes the same hydraulic passage sizes throughout the vanes as is [illegible] in the larger unit. Flow velocities are controlled by altering the number of passages and orifices (including the fuel-rich hot gas orifices between the vanes). Table CIX lists the resulting major injector dimensions and operating parameters. The similarity to the 25K injector is shown pictorially in Figures 311 and 312.

The 10K injector pressure schedule is shown in Table CX. On a dimensionless basis, with chamber pressure as the reference, this is the same schedule as used for the larger engine. This is desired for flow control, which automatically occurs when injecting gases. The change in density with pressure alters the fluid density, the velocity head, and the dimensional pressure loss schedule, as desired.

c. Thrust Chamber

The 10K engine thrust chamber shares its materials and type of construction, as well as cooling scheme and overall configuration with that of the 25K engine. It was found necessary to reduce the thrust chamber pressure from 1800 to 1250 psia to duplicate the chamber low cycle fatigue life. This was caused by the hydraulic dissimilarity of the smaller chamber. The wall heat flux could only be reduced by lowering the gas-side film coefficient through a reduction of gas pressure. With the thrust and chamber pressure given, the throat area, and other dimensions were easily determined. These are listed in Table CXI and on Figure 313 as design specification and design concept picture, respectively.

TABLE CIX

10K MAIN INJECTOR BASIC DESIGN SPECIFICATIONS*

Baseline Engine

Number of Vanes		48
Number of Baffles		8
Number of Orifices		608
Orifice Shape and Size, in. Rectangular, in.		0.0187 X 0.0374
Type and Number of Elements:		
Impinging Doublets	280	
Showerhead	48	
Total Elements		328
Doublet Impingement Angle, Degrees		60
Doublet Impingement Distance, in.		0.052
Staggered Doublet Centerline Spacing, in.		0.093
Vane Centerline Spacing, in.		0.193
Thrust per Element, Lbf.		30.5**
Fuel Rich Hot Gas Injection Velocity, ft/sec		575
Oxidizer Injection Velocity, ft/sec		415
Fuel Rich Hot Gas Injection Temperature, °R		1565
Oxidizer Injection Temperature, °R		550
Injector Weight, lb.		12.40
Axial Vane Length, in.		3.5
Injector Face Diameter, in.		3.0
Oxidizer Pressure Drop, psi		420
Fuel Rich Hot Gas Pressure Drop, psi		110

*Data for 10K lbf thrust at 6.0 Engine Mixture Ratio

**Based on total number of elements:

Thrust/Element, F/E = 35.7 lbf based on doublet elements only.
F/E = 32.9 lbf based on 1/2 of the total number of orifices.

TABLE CX

10K INJECTOR PRESSURE SCHEDULE

Baseline Engine

FUEL RICH HOT GAS INJECTOR CORE LOSS	ΔP, PSIA	OXIDIZER INJECTOR CORE LOSS	ΔP, PSIA
Inlet Manifold	30	Manifold Vane, Total	50
Distribution Plate #1	15	Injector Vane Inlet	10
Distribution Plate #2	15	Heating Channel Friction	100
Inter-Vane Friction Loss	40	Orifice Inlet Plenum	60
Injector Velocity Head	10	Injector Orifice	200
TOTAL LOSS	110		420

TABLE CXI

10K COMBUSTION CHAMBER BASIC DESIGN SPECIFICATION

Throat Diameter, in.	2.26
Chamber Contraction Ratio	
Combustion Zone	2.0
Overall Incl. Injector	2.4
Chamber Shape	Conical
Chamber Half-Angle, Degrees	4.2
Chamber Exit Area Ratio	5.3
Combustion Length, L', in.	6.5
Overall Chamber Length, in.	8.4
Primary Cooling Method	Hydrogen Regenerative
Flow Scheme	Single Pass, Counter Flow
Number and Type of Coolant Channels	91 Rectangular
Gas-side Wall Thickness, in.	0.030 Constant
Channel Depth	Continuously Variable
Channel Width	Stepped, 3 Widths
Land/Channel Width Ratio at Throat	1.0
Channel Height/Width at Throat, in/in	.064/.040
Chamber Inner Wall Material	Zirconium Copper
Hoop Stress Support Method	Wire-wrapped
Axial Load Support Method	External Conical Shell
Thrust Chamber Weight, lbm (to ϵ = 5.3:1)	23.6 w/o clevises

TABLE CXII
10K NOZZLE BASIC DESIGN SPECIFICATIONS

Inlet Area Ratio	5.3:1
Exit Area Ratio	400:1
Exit Diameter, in.	45.2
Length, in.	61.2
Contour	Minimum length Rao optimum
Construction Type	Round Tubular-Furnace Brazed
Tube Wall Thickness, in.	0.010, Constant
Tube Material	ARMCO 22-13-5, Tapered Tubes
Number of Tube Bifurcation Planes	2
Location of Bifurcations	ε = 25 and ε = 200
Coolant Scheme	Two-pass Hydrogen Regen.
Bifurcation Joint Type	Circular Tube Surrounding Back-to-back "D" Tubes, Brazed
Turnaround Manifold Type	"U" Tubular @ ε = 400, Brazed
Number of Tubes:	Total Segments = 497
ε = 6 to ε = 25	71
ε = 25 to ε = 200	142
ε = 200 to ε = 400	284
Number of Stiffening Rings	5
Attachment to Thrust Chamber	Rebrazable Joint
Nozzle Weight, lbm (ε = 5.3:1 to 400:1)	72.2

TABLE CXIII

10K PREBURNER BASIC DESIGN SPECIFICATIONS*

Injector Type and Material		Brazed Platelet, N_i-200
Propellant Injection Phase		Gas/Gas
Injector Pattern		Like-on-Like Doublets
Number of Orifices:		
Fuel		2520 → 1260 Doublets
Oxidizer		1274 → 637 Doublets
Total	=	3794 → 1897
Orifice Shape and Size		Rectangular
Fuel, in. X in.		0.010 X 0.020
Oxidizer, in. X in.		0.010 X 0.020
Doublet Impingement Angle		
Fuel & Oxidizer, Degrees		60
Impingement Distance from Face, In.		
Fuel, in.		0.030
Oxidizer, in.		0.055
Injector and Chamber Dia., in.		2.4
Fuel Injection Velocity, ft/sec		1000
Oxidizer Injection Velocity, ft/sec		125
Fuel Injection Temperature, °R		450
Oxidizer Injection Temperature, °R		400
Total Fuel Pressure Drop, psi		205
Total Oxidizer Pressure Drop, psi		385
Chamber Type		Oxidizer Regen. Cooled, with Fuel Regen. Cooled Liner
Combustion Stability Device		Acoustic Resonator Integral with Chamber and Liner
Chamber Length, in.		9.0
Total Weight, lbm		18.7

*Data for 10K lbf thrust at 6.0 engine mixture ratio.

TABLE CXIV

10K THRUST CHAMBER IGNITER BASIC DESIGN SPECIFICATION

Type	Hot Gas Torch
Initiator	Spark
Electrode Cooling	Submerged in O_2 Flow
Spark Gap Width, in.	0.035
Spark Voltage, Kv	20
Spark Rate Sparks/sec.	50
Spark Energy, Millijoules/Spark	5
Torch Mixture Ratio, O/F	1.5
Torch Temperature (minimum) °F	1100
Fuel Flow Rate, lbm/sec	0.022
Oxidizer Flow Rate, lbm/sec	0.033
Duration of Operation, Sec/Engine Start	0.75
Materials:	
Housing	ARMCO 22-13-5
Chamber Liner/Flame Tube	Haynes 188
Electrical Seal	Brazed/Ceramic
Injector Head	Zirconium-Copper
Total Weight, lbm (with Exciter)	12.1

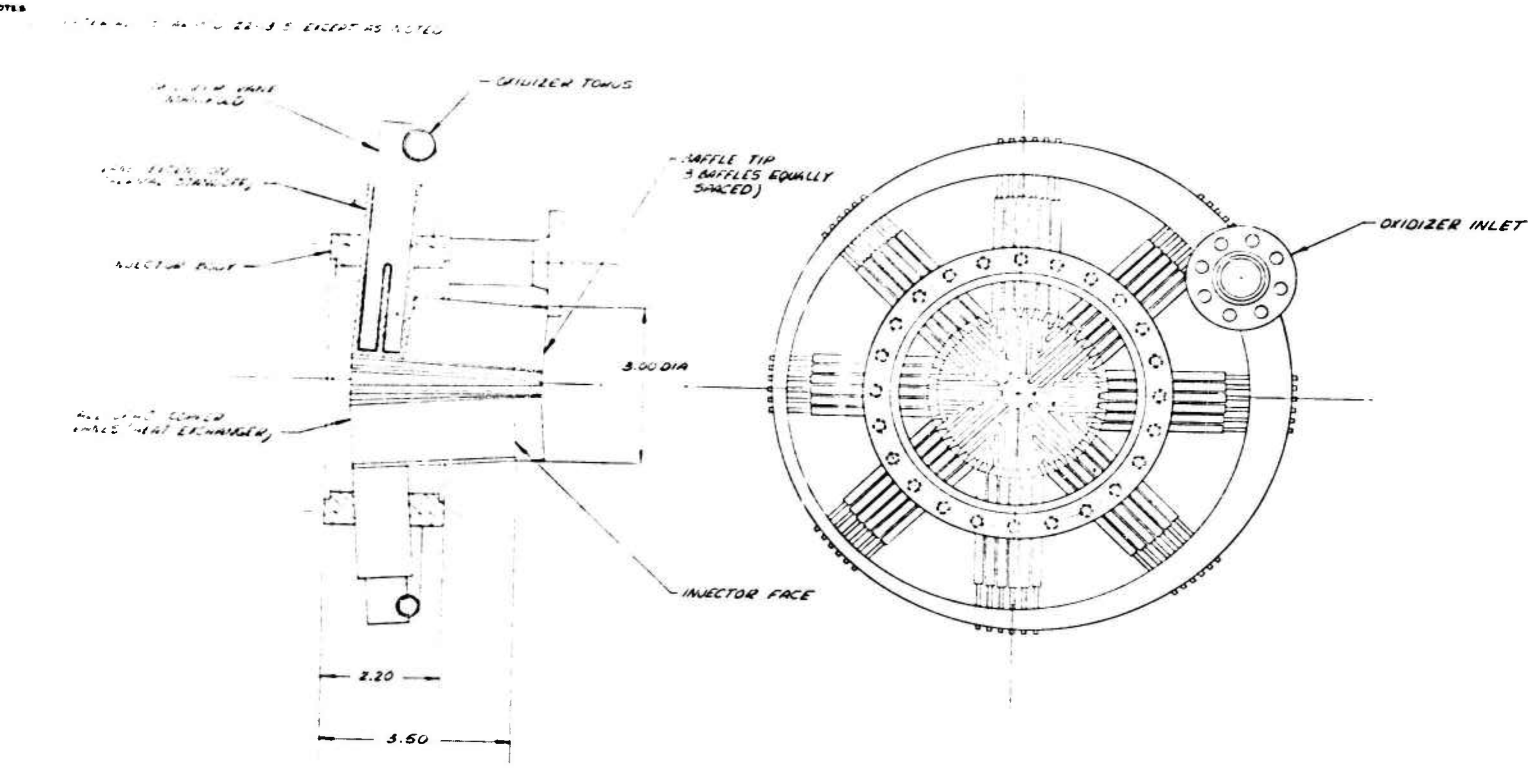


Figure 311. 10K Injector

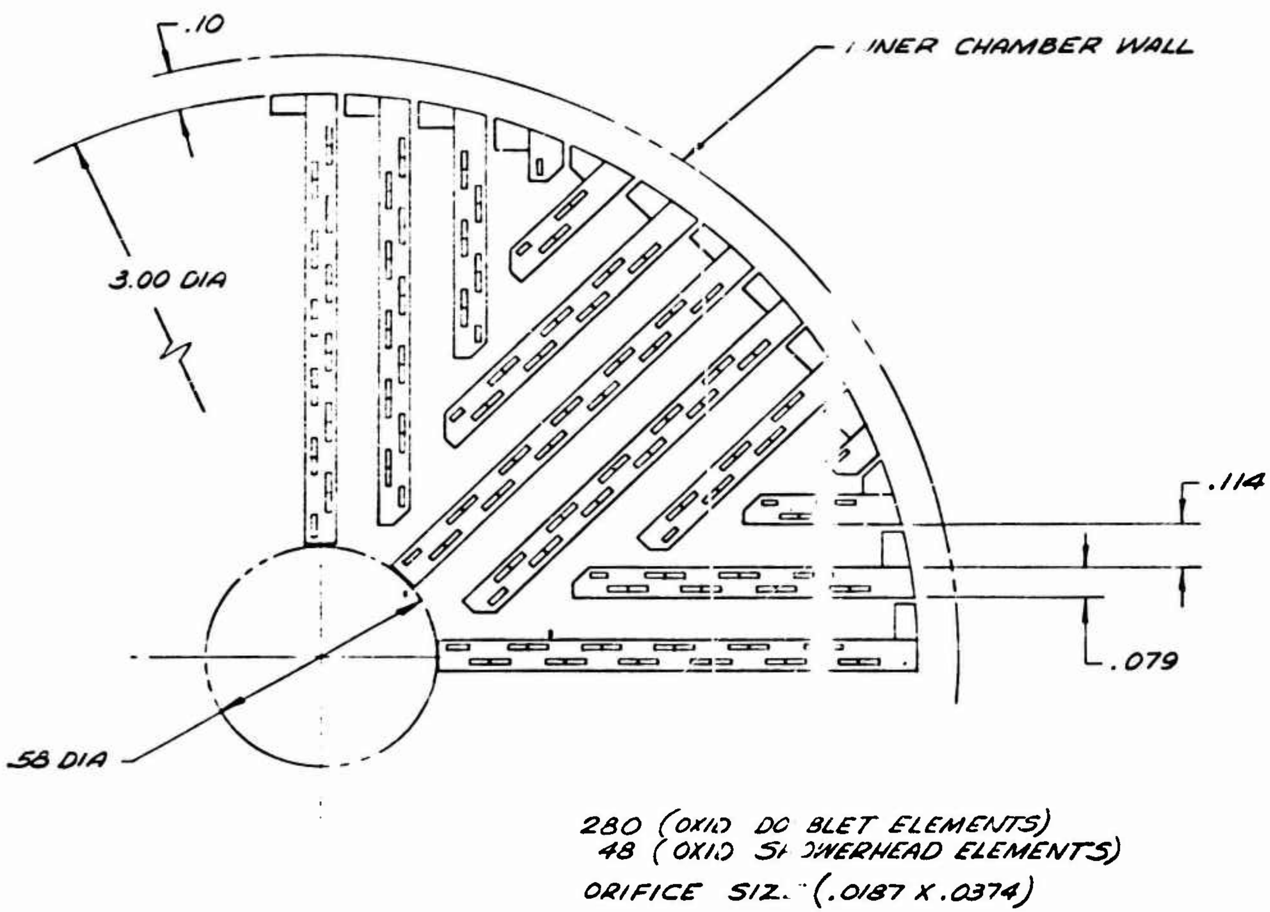


Figure 312. 10K Injector Face Details

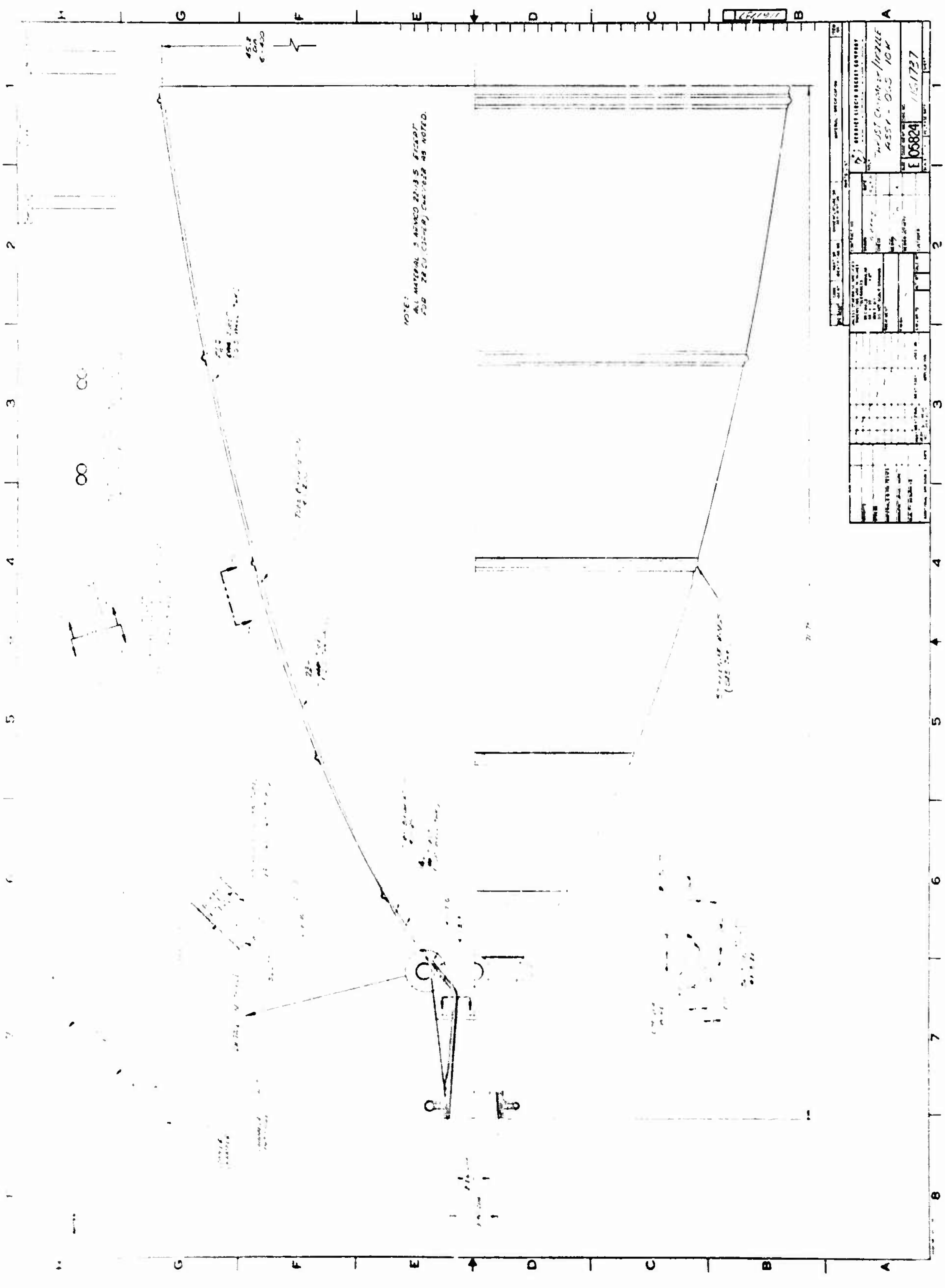


Figure 313. 10K Combustion Chamber/Nozzle

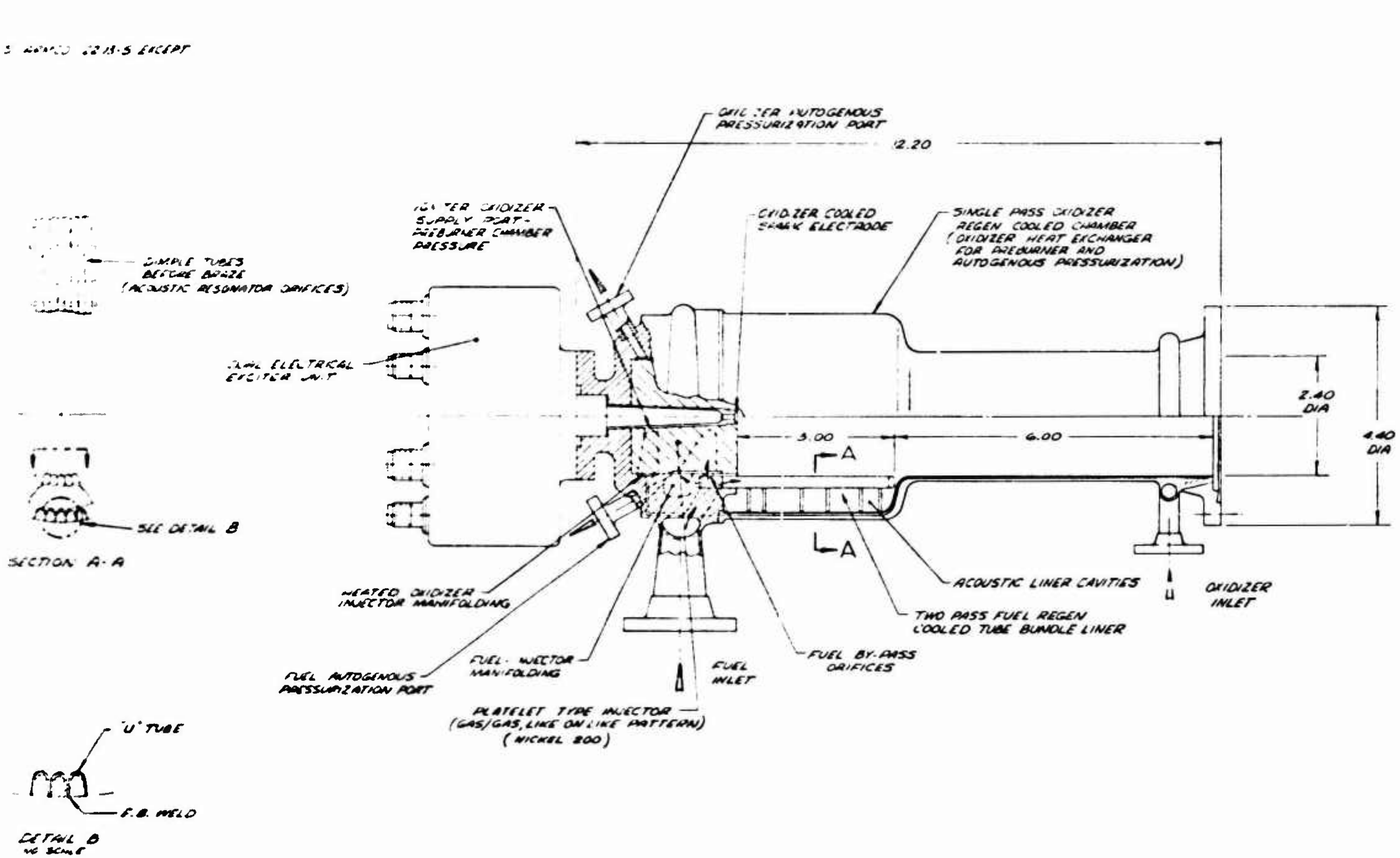


Figure 314. 10K Prebu:rner

The selection of the chamber pressure of 1250 psia is based on the low cycle fatigue life capability of the ZrCu chamber material, which was established experimentally at the ALRC facilities. Since the start of this contract considerable more data was made available from other sources and an effort was made to correlate these data for the two most promising chamber materials.

Zirconium Copper

Silver - Zirconium Copper

The result of this analysis indicates that the low cycle fatigue life is strongly effected by the environmental conditions. The data presented in Figure 315 and Figure 316 indicates the low cycle fatigue for these materials measured in air and in inert environments, indicating the testing in air to result in lower fatigue life than tests in inert environment by a factor of about 3. This fact is attributed to the oxidation within the fatigue cracks, when tested in air.

The real environment in LOX/hydrogen engine is superheated steam and free hydrogen and its effect has not been established to date, but it is speculated that the available data will bracket the actual environmental effects. The conclusion reached from this analysis is that the chamber life estimate based on the data in air is conservative.

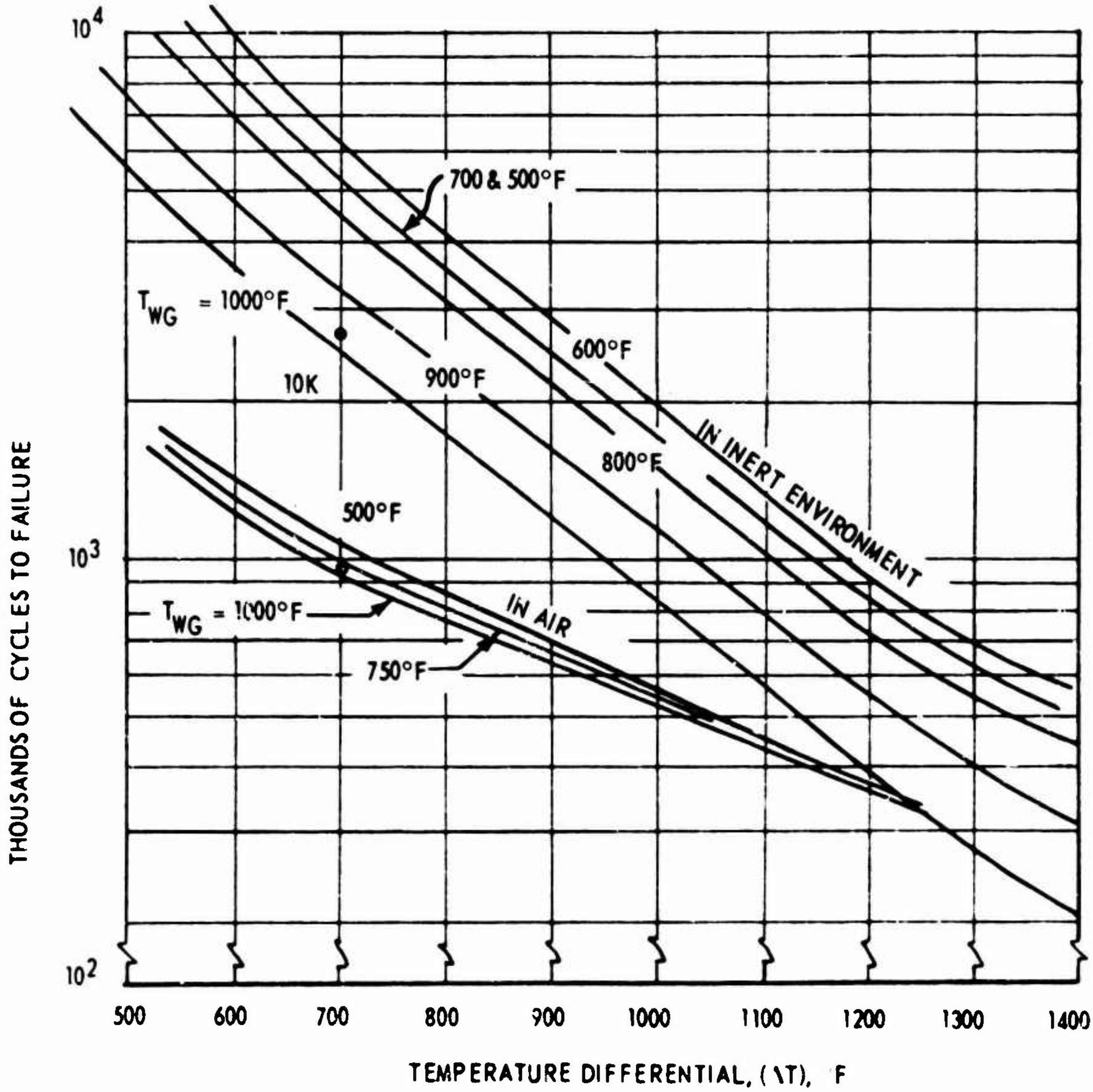


Figure 315. Life Estimates for OOS Chamber Made of Zirconium Copper

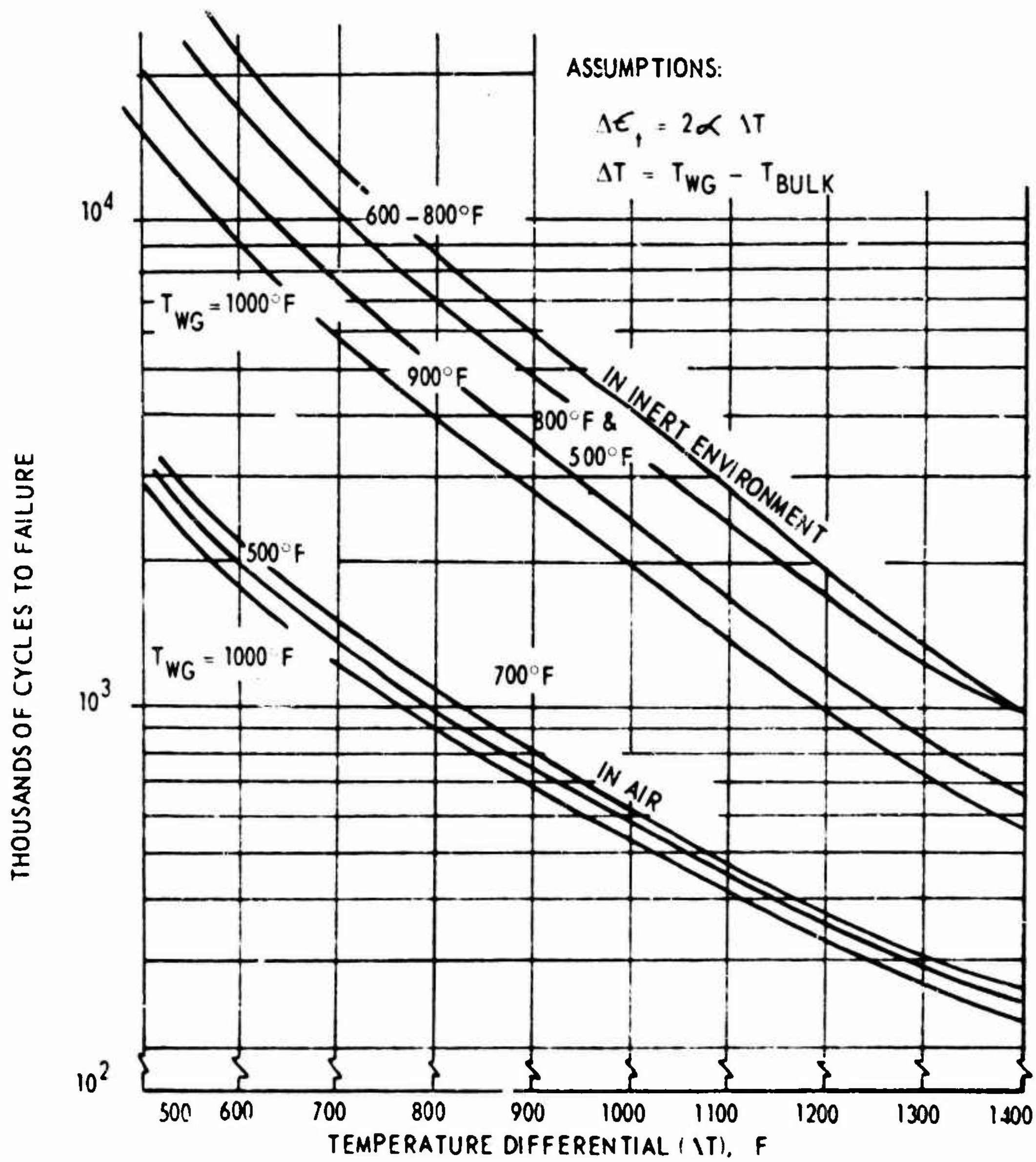


Figure 316. Life Estimates for OOS Chamber Made of Silver Zirconium Copper

d. Nozzle

The expansion nozzle for the 10K engine comprises a large portion of the total engine envelope and weight, more so than was the case for the 25K engine. This is in spite of the fact that the same materials, design type configuration, as well as cooling scheme is used. This is also in spite of the fact that 0.010-in. thick regenerative tube wall thicknesses are used instead of 0.015-in. on the larger thrust engine. The 10K engine nozzle has a larger overall area ratio, and operates at a lower chamber pressure. Therefore, its length and weight are nearly equal to that of the larger nozzle. Since the remainder of the engine is considerably smaller and lighter, the nozzle becomes a component of major importance to vehicle performance. Design specifications and concept are shown in Table CXII and Figure 313, respectively.

e. Thrust Chamber Thermal Characteristics

The thrust chamber assumed coolant passage geometry is shown in Figure 317 for the Zr Cu chamber. This geometry was used to determine the chamber thermal characteristics.

The pressure drop characteristics at full thrust operation is shown in Figure 318 indicating the coolant sensitivity to pump discharge pressure and was used to obtain the feed system power balance. The coolant characteristics for throttling conditions are shown in Figure 319.

The heat transfer analysis had the objective to define the coolant condition for meeting the 300 thermal cycle requirements. The result of this analysis is summarized in Figures 320 and 321 for the throat conditions and also includes the condition at the throat for off mixture ratio conditions at full thrust. Indications are that the chamber has more than adequate life at all operating conditions.

The throttling conditions at MR = 6.0 are described in Figure 322 indicating a rapid increase of chamber life with throttling. This indicates the capability to improve engine cycle life by simply directing the engine to a slightly lower thrust level.

f. Preburner

The 10K engine preburner chamber is the same length and design/construction type as the larger, 25K unit. Since the chamber is a heat exchanger, it, in common with the main injector, utilizes fewer coolant

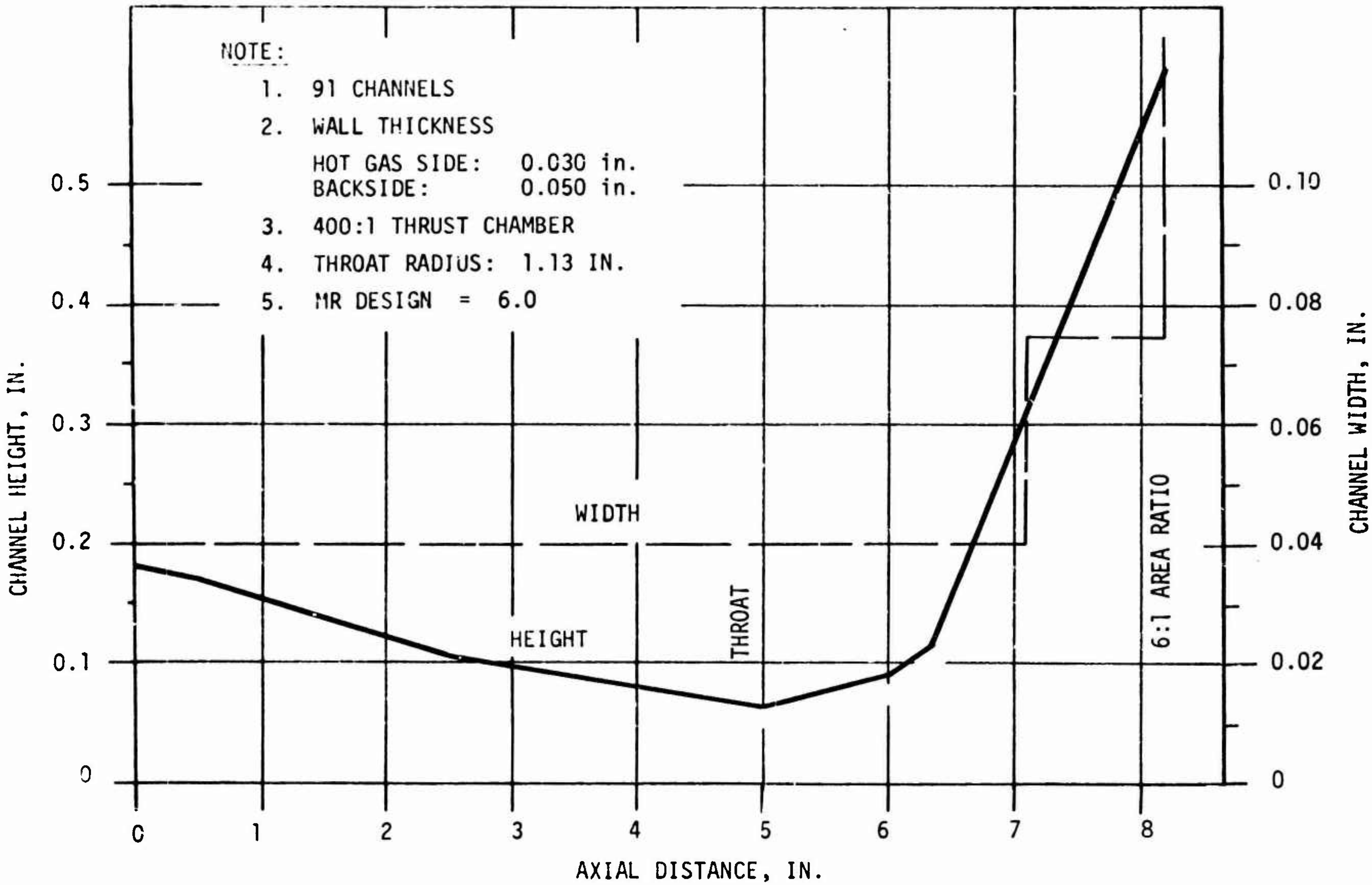


Figure 317. 10K Copper Nozzle Coolant Channel Geometry

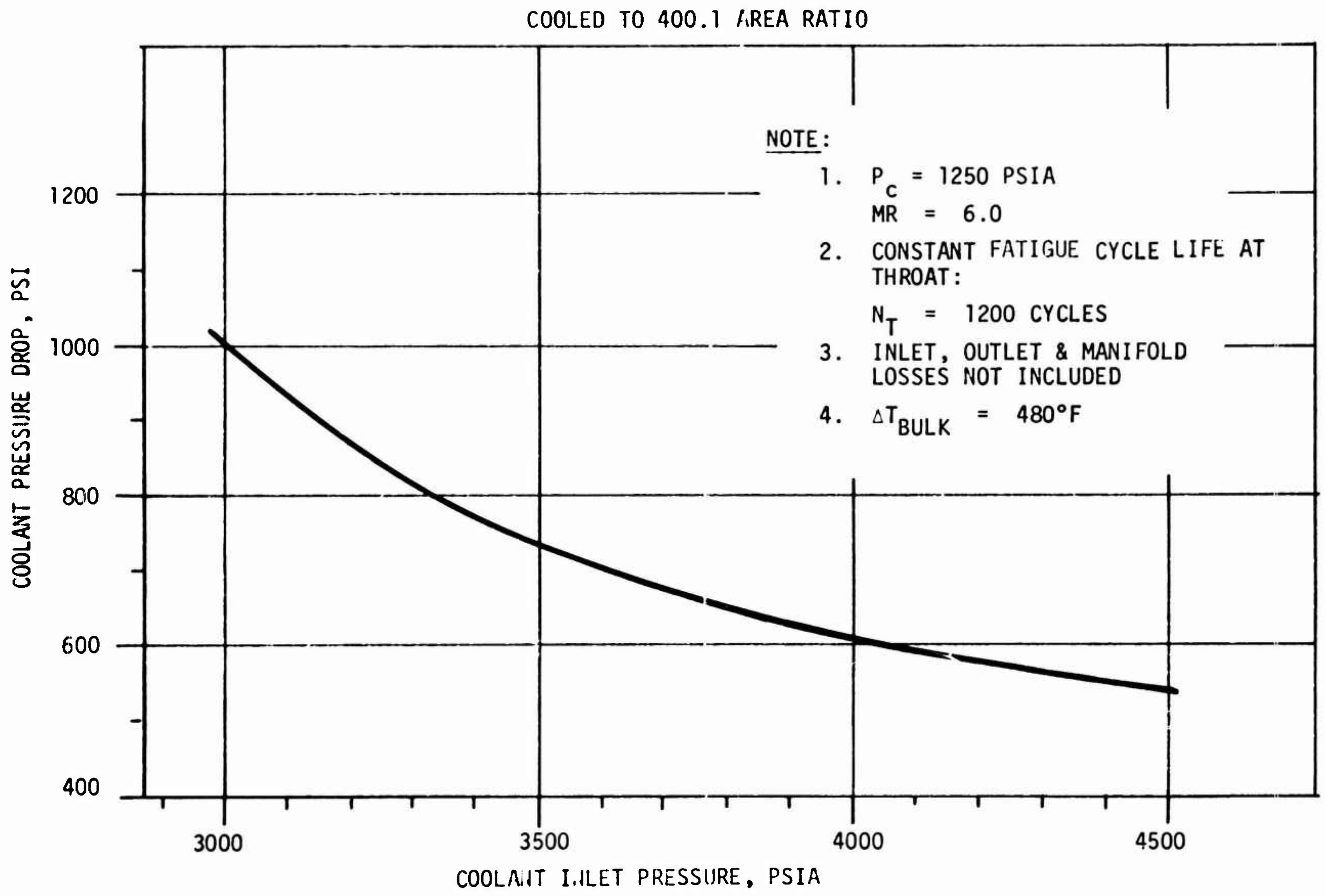


Figure 318. 10K Coolant Characteristics for Steady State Operation

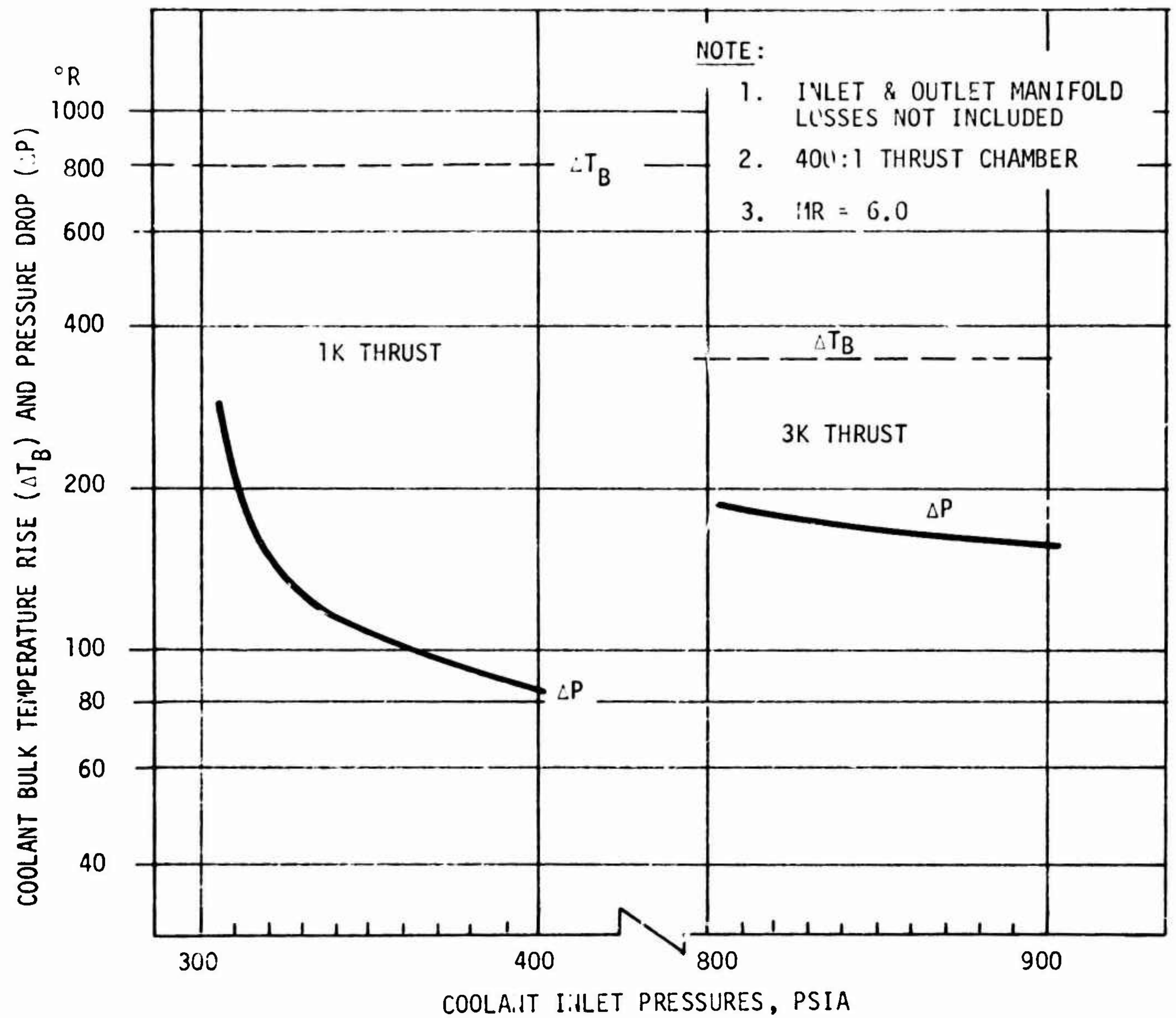


Figure 319. 10K Coolant Characteristics for Throttled Operation

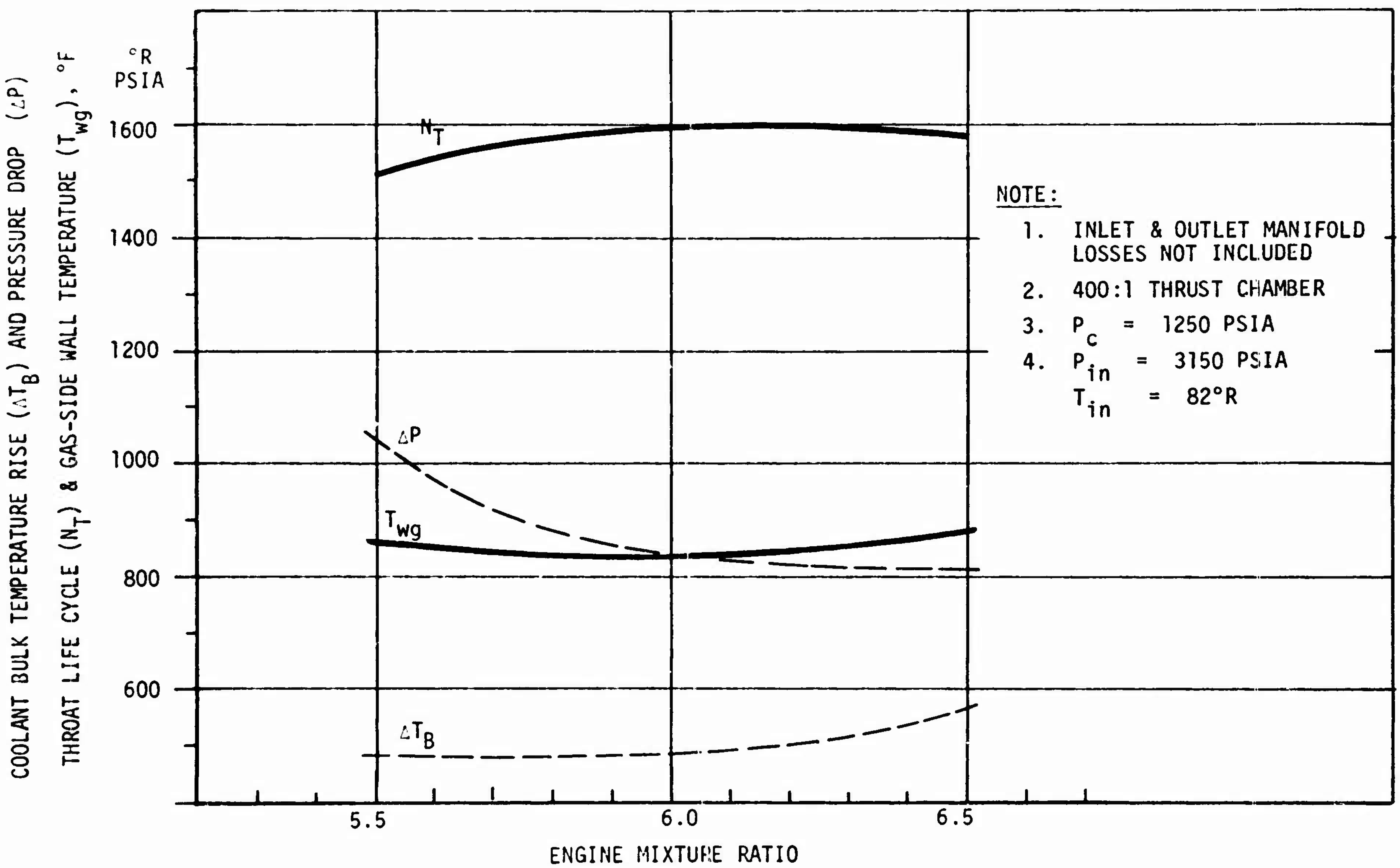


Figure 320. 10K Various Operating Parameters vs Mixture Ratio - Steady State

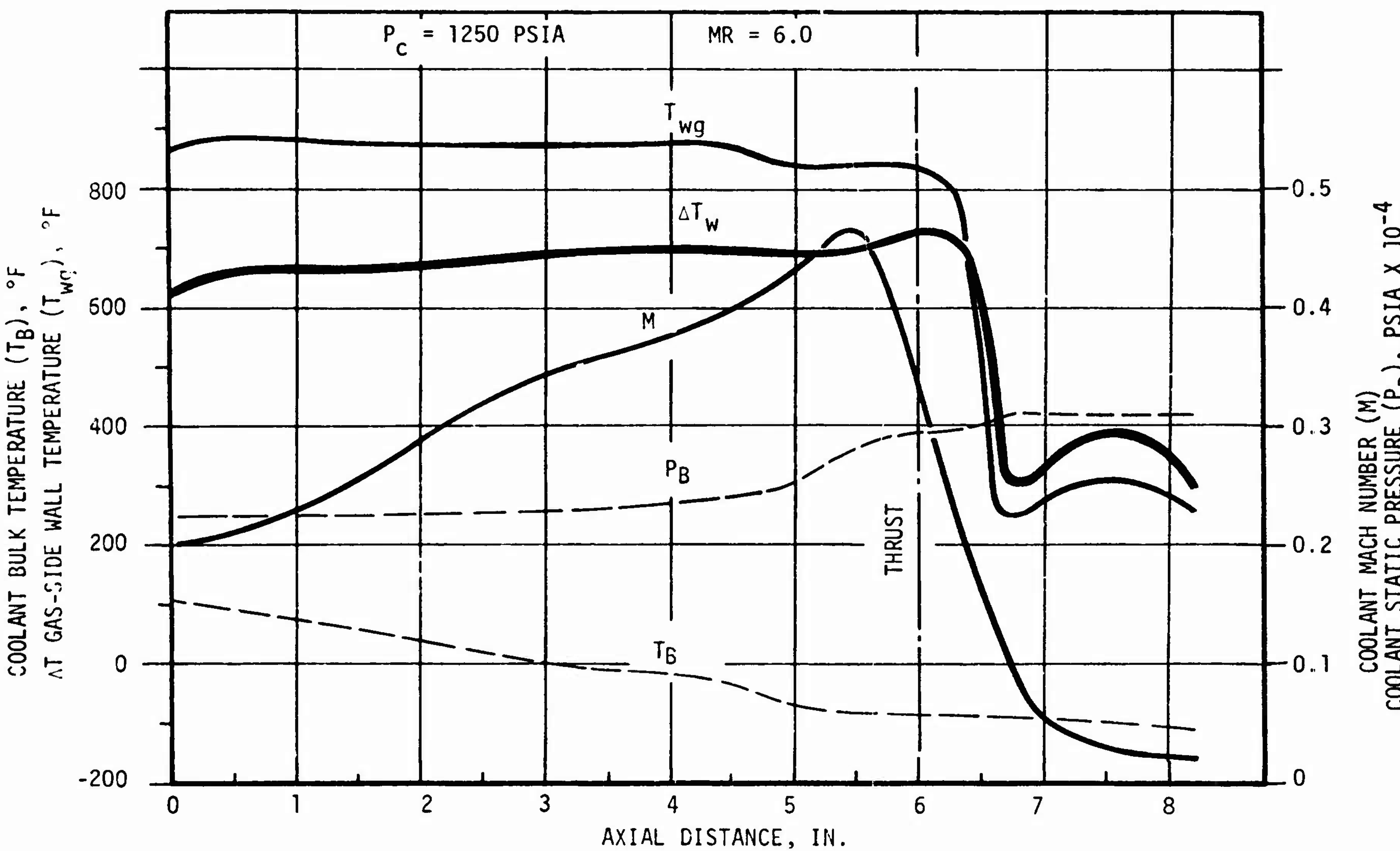


Figure 321. 10K Temperature vs Axial Distance

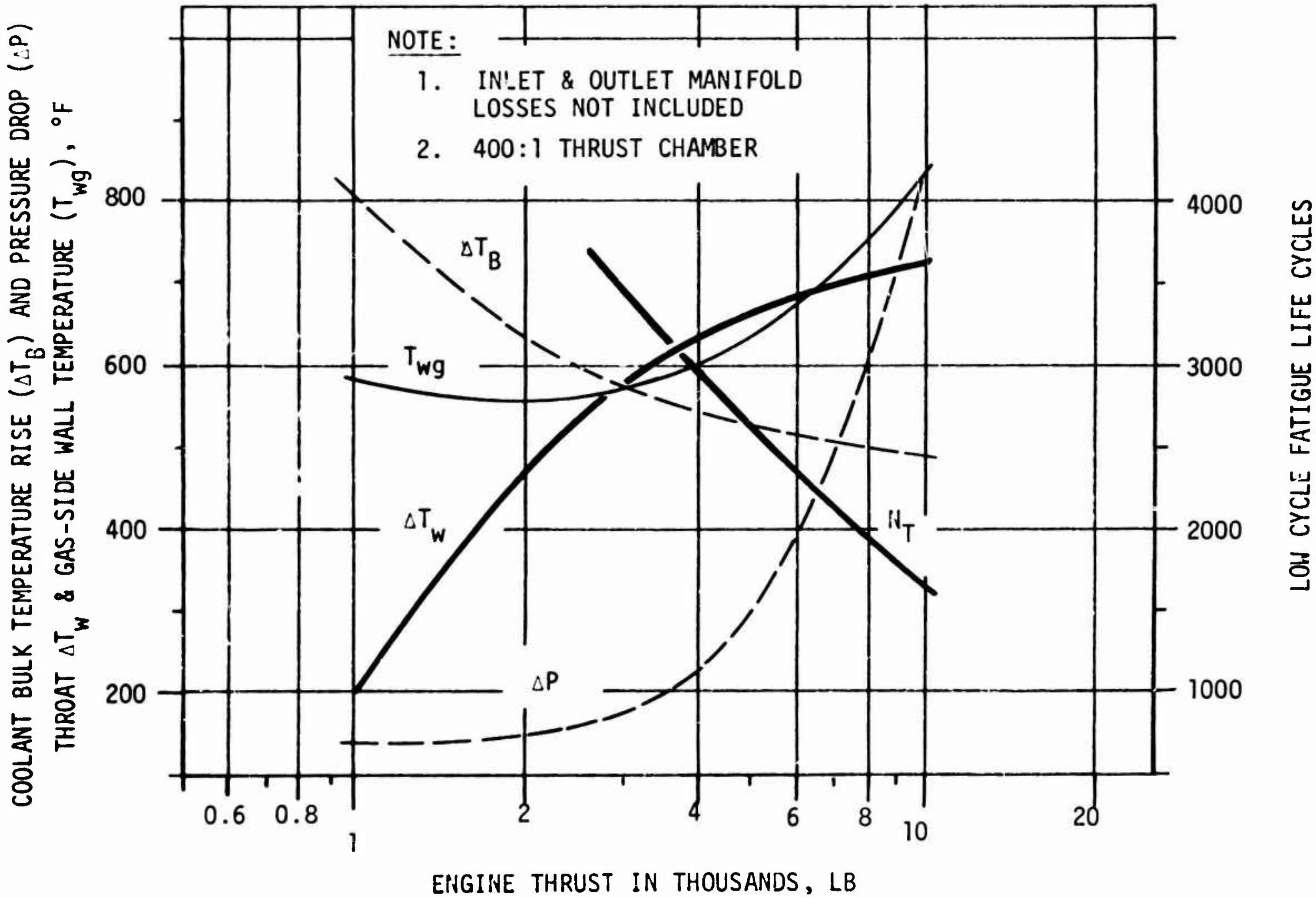


Figure 322. 10K Varicus Operating Parameters vs Thrust - Throttled Condition

passages of similar size to form a chamber of smaller flow area to contain the lower volumetric flow of preburner reactants. The total heat transferred to the oxygen and hydrogen coolants per lb per second is similar, to obtain the same thermal schedule. The injector, again is similar, containing fewer injection orifices, arranged in the same pattern as shown for the 25K preburner injector. The preburner basic design specifications are given in Table CXIII, and depicted pictorially in Figure 314.

g. Igniter

The igniter for the 10K engine main injector is identical to that of the 25K engine in every respect. The reasons for this are that the ignition requirements are the same, both within the igniter chamber and in the thrust chamber. This is because the same propellants, mixture ratios, and start pressure schedules require the same igniter diameter and total heat generation rate. The basic design specification for the main igniter is shown in Table CXIV.

The 10K engine preburner igniter operates in the same manner as its larger counterpart. Therefore, the same exciter, electrode, and feed system is utilized in both preburners.

h. Turbopumps

(1) Requirements

The operational requirements for the 10,000 lbf vacuum engine are identical to those of the 25,000 lbf engine with appropriate adjustments for the 10K flow and pressure schedule.

(2) Design Selection

(a) Design Criteria

The turbopumps for the 10K engine are based on the same structural criteria as used for the 25K turbopump. This criteria is summarized below:

1 P mp Impeller

1600 ft/sec impeller maximum rated tip speed (Titanium) to achieve 300 thermal cycles.

2 Turbine Rotors

1860°R turbine inlet temperature to achieve 300 thermal cycles.

1300 ft/sec turbine mean blade speed to achieve a disk design life of 10 hours. The burn mixture consists of one 1000 second long burn plus eleven 40 second long burns plus eleven 40 second short burns.

Turbine blade root stress allowable value of 31000 psi (50% of 10 hour creep rupture strength).

The design parameters used for the 10K turbopump are given in Table CXV and differ from the 25K design parameters in the areas tabulated below:

	Oxid.		Fuel	
	25K	10K	25K	10K
Max. RPM Ratio, Main/Low Speed	3.5:1	5:1	3.5:1	5:1
Bearing DN - Low Speed Pump	215000	126000	343000	24000
Number Stages - Main Pump	1-1/2	1-1/2	3	2
Shaft Speed - Main Pump	50000	63000	80000	100000
Internal Recirculation Allocated	5%	7.5%	7%	15%
Turbine Bypass, Allocated	4%	17.5%	4%	6%
Turbine End Bearing DN - Main	1.5×10^6	1.2×10^6	2×10^6	2×10^6

The relative shaft speeds of the low speed pump were reduced for the 10K design to reduce their power requirements. The 10K fuel pump is designed with two centrifugal stages compared to the three stages for the 25K TPA design. The lower system pressure permitted the lower head generation with two stages while maintaining the impeller tip speed below 1600 ft/sec. The 10K TPA design point shaft speed values were increased. (Refer to the section below for design speed selection.) The pump allocated recirculation flow and the turbine bypass flow values were increased from the 25K values because as pump sizes are reduced, the leakage area does not reduce

TABLE CXV

10K ENGINE

TPA DESIGN PARAMETERS

	Oxid	Fuel
Low Speed Pump Assembly		
NPSH, feet	16	60
Thermodynamic Suppression Head, ft	Calc.	Calc.
Min. Ratio of Effect. NPSH, Run/Breakdown	1.87	1.87
Suction Specific Speed, Breakdown	45000	45000
Max. RPM Ratio, Main/Boost	5:1	5:1
Specific Speed, Maximum	4000	4000
Suction Diameter Ratio, Hub/Tip	0.4	0.3
Tip Diameter Ratio, Exit/Suction	1.0	0.916
Mean Diameter Ratio, Exit/Suction	1.1	1.054
Turbine Drive	FFH*	FFH*
Ratio Bearing Spacing/Shaft Diameter	2.17	2.17
Ratio Turbine Overhang/Shaft Diameter	3.17	3.17
Bearing Dn.	126,000	24,000
Main High Speed Turbopump Assembly		
High Speed Inducer		
Thermodynamic Suppression Head, ft	Calc.	Calc.
Min. Ratio of Effect. NPSH, Run/Breakdown	1.87	1.87
Suction Specific Speed Breakdown	30000	30000
RPM Ratio, Inducer/Main	1:1	1:1
Specific Speed, Maximum	4000	4000
Suction Diameter Ratio, Hub/Tip	0.512	0.515
Tip Diameter Ratio, Exit/Suction	1.0	1.0
Mean Diameter Ratio, Exit/Suction	1.1	1.1
High Speed Main Pump		
Thermodynamic Suppression Head, ft	Calc.	Calc.
Min. Ratio of Effect. NPSH, Run/Breakdown	2.27	2.50
Suction Specific Speed Breakdown	10000	10000
Shaft Speed	63,000	100,000
Number Stages	1-1/2	2
Internal Recirculation, Allocated, %	7.5	15
High Speed Main Turbine		
Type	Axial	Axial
Number Stages	1	2
Energy Extraction Means	Impulse	PI**
Inlet Temperature, °R	1860	1860
Mean Blade Speed, ft/sec	1100	1300
Nozzle Angle	15	20
Max. Diameter Ratio, Hub/Tip	0.90	0.90
Min. Diameter Ratio, Hub/Tip	0.85	0.85
Inlet Manifold Mach No.	0.3	0.3
Exit Manifold Mach No.	0.5	0.5
Turbine Bypass, Allocated, %	17.5	6
Power Transmission		
Turbine End Bearing Type	Dplx Ball	Dplx Ball
Turbine End Bearing DN	1.26×10^6	2×10^6
Pump End Bearing Type	Dplx Ball	Dplx Ball
Pump End Bearing DN	1.07×10^6	2×10^6

*FFH - Full Flow Hydraulic
**PI - Pressure Compounded, Impulse

proportionately. The particularly large increase in the fuel pump recirculation resulted from the miniaturization affects and from reducing the number of pump stages. The reduction of pump stages increased the pressure drop across the hydrostatic seals which in turn increased the leakage flow.

(b) Shaft Speed Determination

The design point shaft speed for the fuel main turbopump was established at 100,000 RPM. The relationship of allowable pump shaft speed and TPA weight to available NPSH are shown in Figure 323. The fuel TPA shaft speed was established at 100,000 RPM to limit turbine blade root stresses to 31,000 psi. Increasing shaft speed above 100,000 RPM would reduce the turbine diameter. This action would require an increased blade length which would increase turbine blade root stress above the limiting 31,000 psi value. Pump operation at 100,000 RPM requires 123 feet NPSH and results in a weight value of 21.5 pounds. This NPSH value exceeds the specified minimum of 60 feet. Pump operation with 60 feet NPSH would permit a maximum pump shaft speed of 81,000 RPM with a resulting turbopump weight of 44.5 pounds. A low speed pump weighing 1.8 pounds will permit the 100,000 RPM main pump design operating point giving a combined boost pump plus turbopump weight of 23.3 pounds as compared to a 44.5 pound TPA without a low speed pump. The weight values noted above were obtained from a computerized TPA weight program. These computed values while differing from the values computed for the actual TPA design, do provide meaningful weight trends for trade studies.

The oxidizer TPA shaft speed was established at 63,000 RPM. This value was selected to equalize the shaft spin up time since the oxid and fuel turbopumps operate from a common gas generator. Increasing the design shaft speed would reduce the diameter of the turbine which in turn would reduce its rotor moment of inertia (turbine rotor is the predominate inertia component) and reduce the spin up time. The turbine mean blade speed was established at 1100 ft/sec, a compromise value to achieve the desired spin up times, and to adjust turbine rotor weight to achieve shaft critical speed values that fall between the 2nd and 3rd critical speeds modes.

The relationship of oxid pump allowable shaft speed and resulting TPA weight to available NPSH are shown in Figure 324. Pump operation at 63,000 RPM requires 65 feet NPSH and results in a weight value of 16 pounds. This NPSH value exceeds the specified minimum of 16 feet. A boost pump weighing 3.5 pounds will permit the 63,000 RPM main pump design operating point giving a combined boost pump plus turbopump weight of 19.5 pounds as compared to a 153 pound TPA weight without a boost pump.

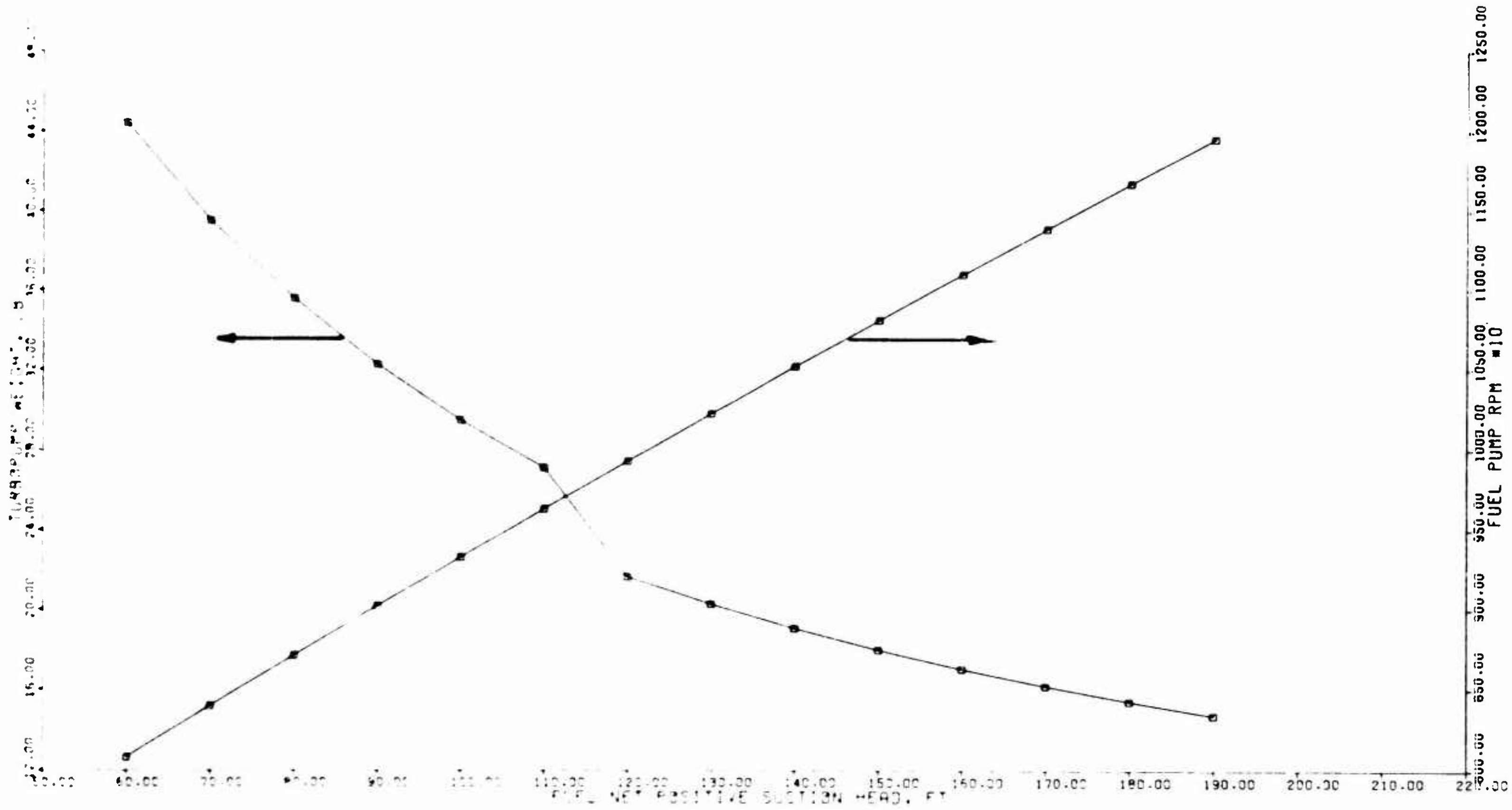


Figure 323. 10K Engine Pump Speed and Turbopump Weight vs Suction Head - Fuel

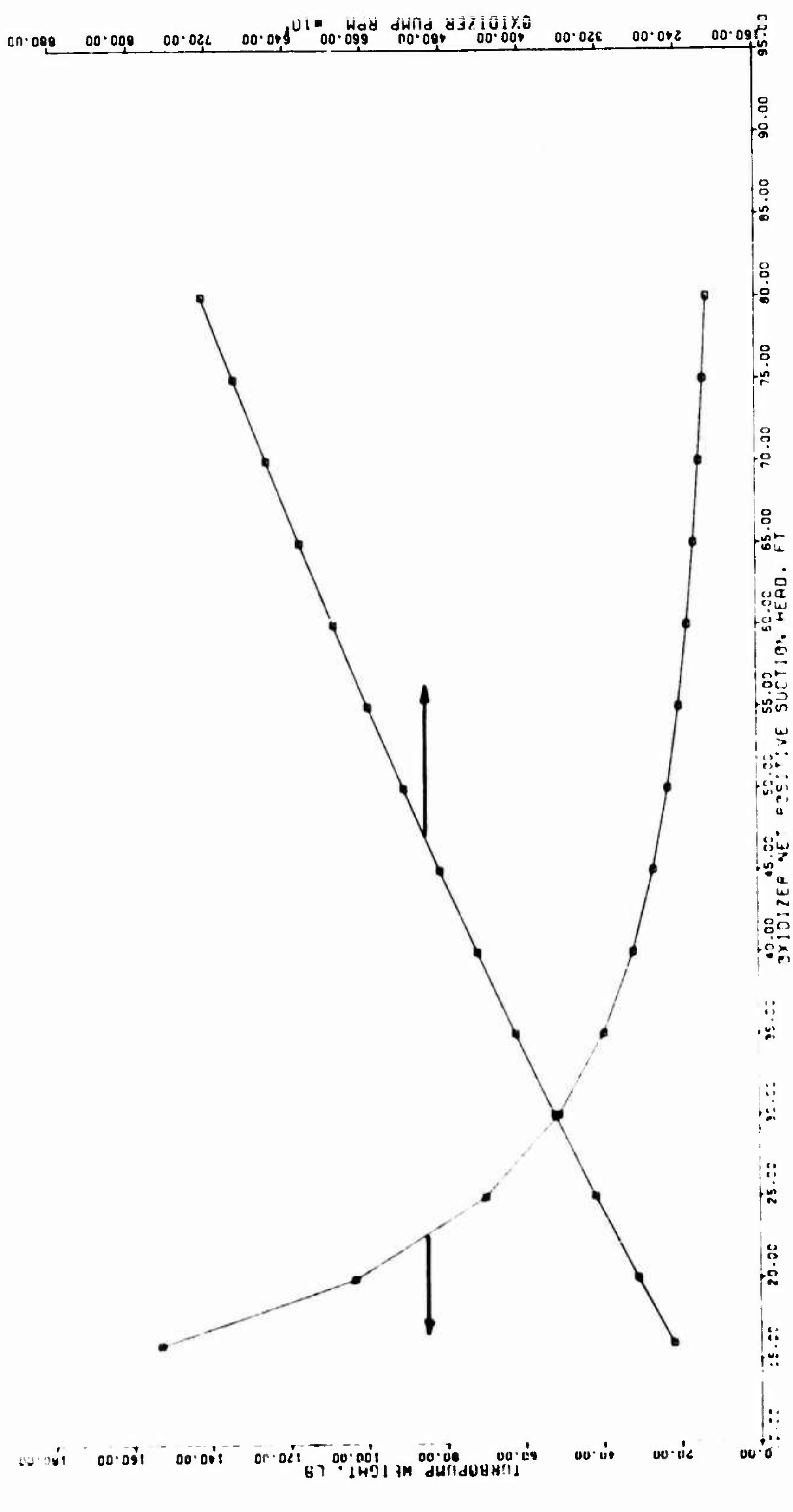


Figure 324. 10K Engine Pump Speed and Turbopump Weight vs Suction Head - Oxidizer

(c) Fuel TPA Candidate Configuration

Six configurations of the fuel turbopump were considered in the fuel TPA selection. These six candidates are shown by conceptual sketches in Figures 325 through 330. A turbopump with a gear driven oxidizer pump was analytically evaluated where the oxidizer pump is driven by the fuel pump turbine through a speed reducing gear box. The analysis showed that a slight increase in cycle efficiency would be achieved as shown in Figure 299 by a slightly lower power balance pressure (fuel pump discharge pressure). This configuration was selected as an alternate configuration and is presented in Figure 330. From the six TPA candidates, Figures 325 through 329, concept No. V, (Figure 328) was selected. The selected concept positions the shrouded impeller front to back with the pump end bearing located between the two impellers. This concept was selected on the basis that it:

1 Eliminated the tight axial clearance requirement.

2 Permitted the high speed inducer to be driven by the impeller shroud (the full flow hydraulic turbine drive concept necessitates a shroud driven high speed inducer), and

3 Permitted a first stage impeller design with a low hub to tip diameter ratio.

The selected concept has the disadvantage that LH_2 will leak past the hydrostatic seal from the high pressure area of the first stage impeller to the suction of the first stage. This leakage flow will be heated as a result pump inefficiency and where this leakage flow enters the low pressure area of the 1st stage suction, some propellant could flash to vapor. Therefore, design consideration might be given to (1) adjusting pressure values where the leakage flow re-enters the main stream and to (2) adjusting the through flow areas to accommodate the increased volume flow resulting from the presence of vapor.

The six candidate TPA concepts including the selected concept No. V, are compared and rated from 1 (selected) to 6 (least attractive) in Table CXVI. Concept VI (front to back unshrouded impellers - bearing between stages) was a close second and was rated No. 2 on the basis that the unshrouded impeller concept did not lend itself to the full flow turbine driven boost pump drive in that it does not have a shroud to drive the high speed inducer.

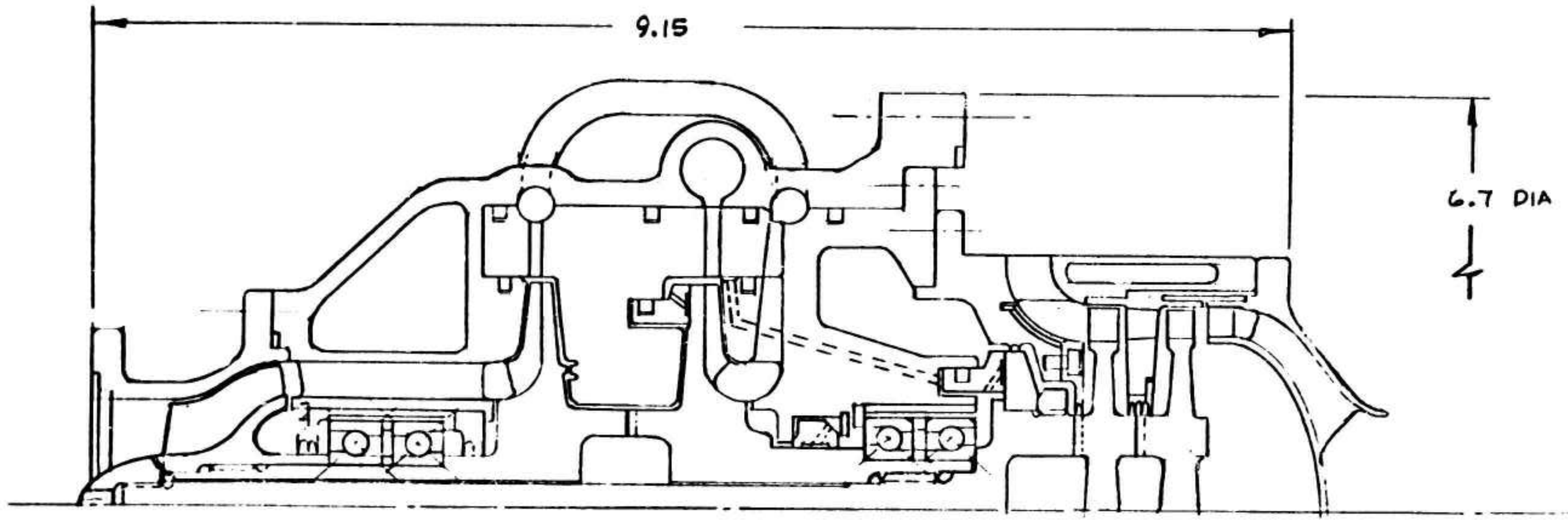


Figure 325. 10K Fuel Turbopump Concept, I - Back-to-Back Unshrouded Impellers

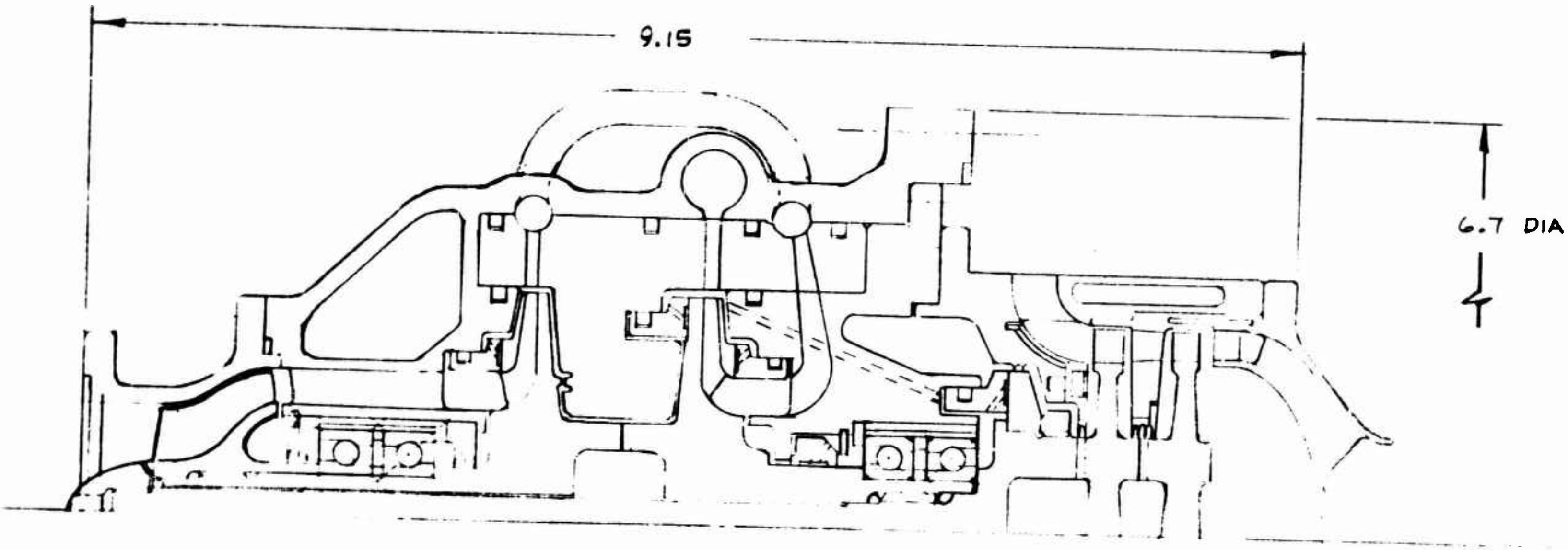


Figure 326. 10K Fuel Turbopump Concept, II - Back-to-Back Shrouded Impellers

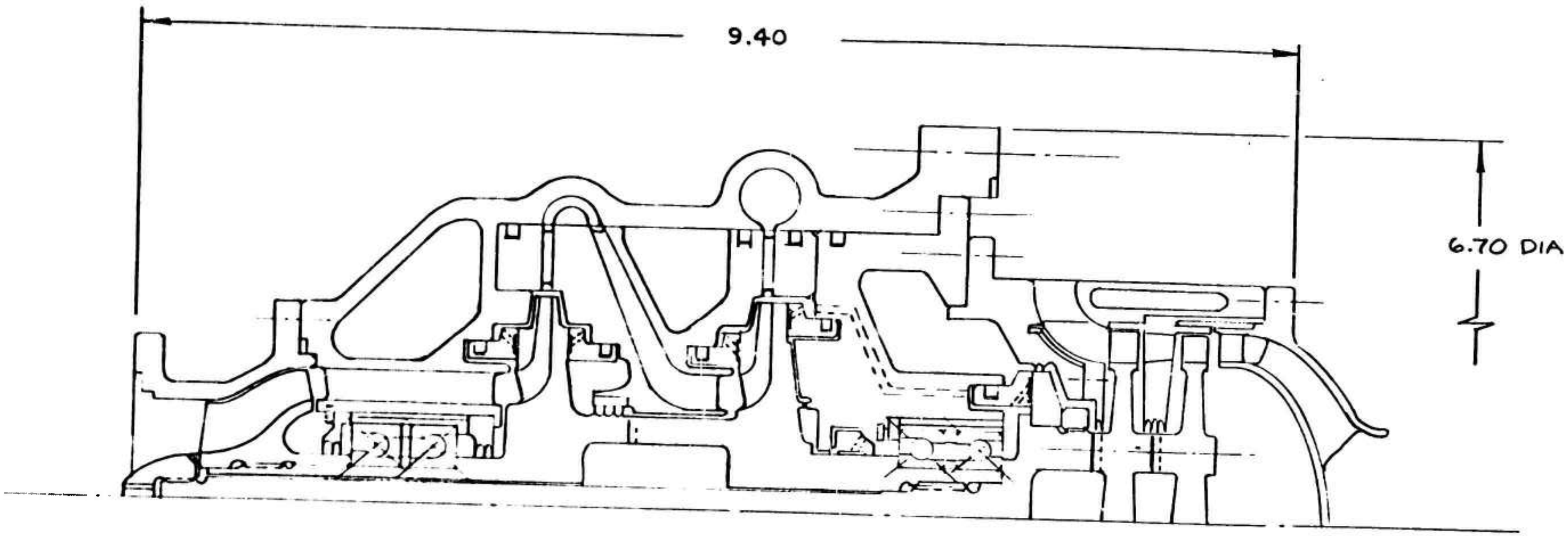


Figure 327. 10K Fuel Turbopump Concept, III - Front-to-Back Shrouded Impellers

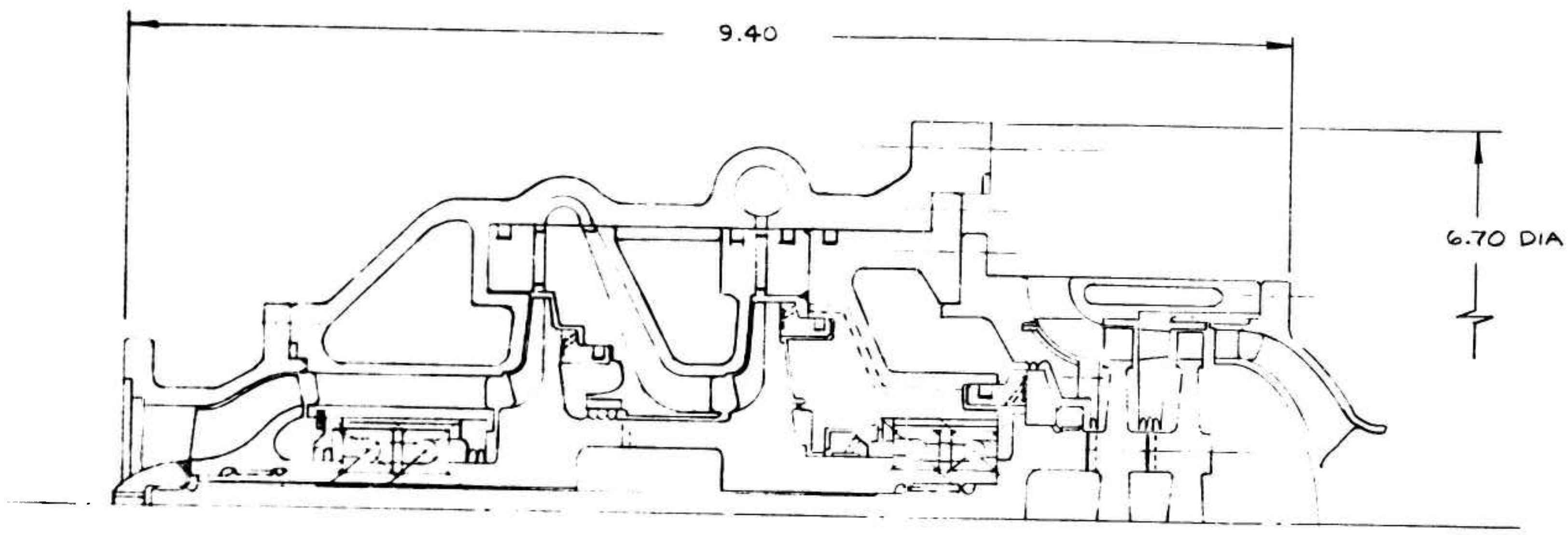


Figure 328. 10K Fuel Turbopump Concept, IV - Front-to-Back Unshrouded Impellers

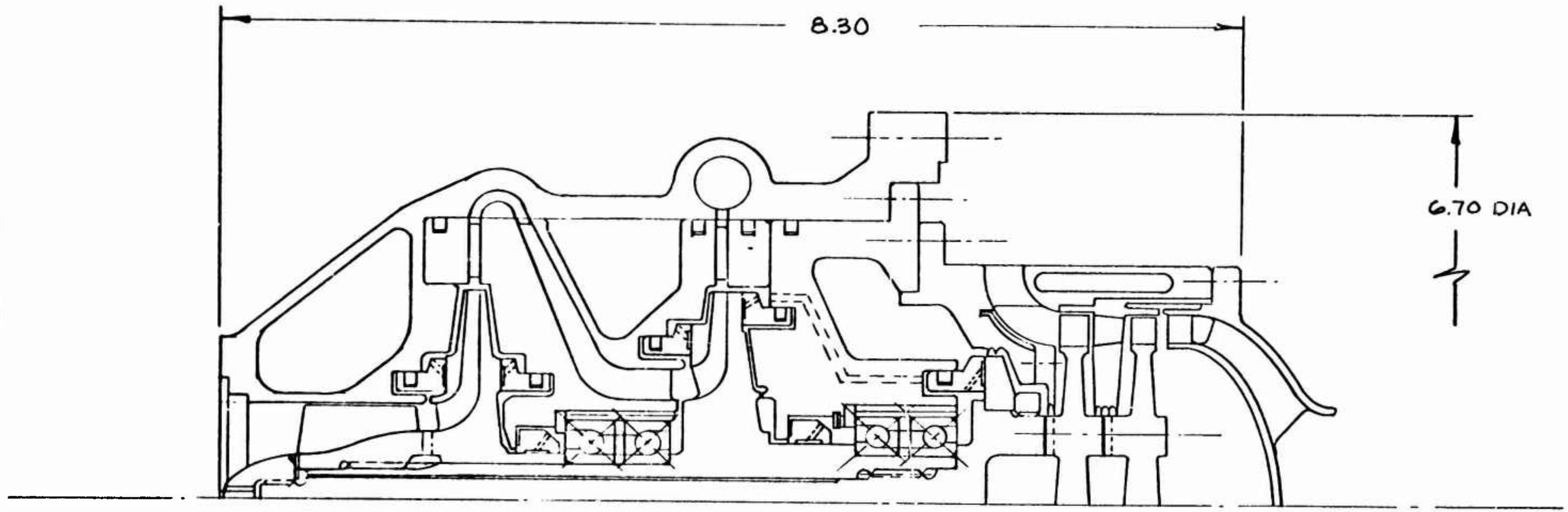


Figure 329. 10K Fuel Turbopump Concept, V - Front-to-Back Shrouded Impellers - Bearings between Stages

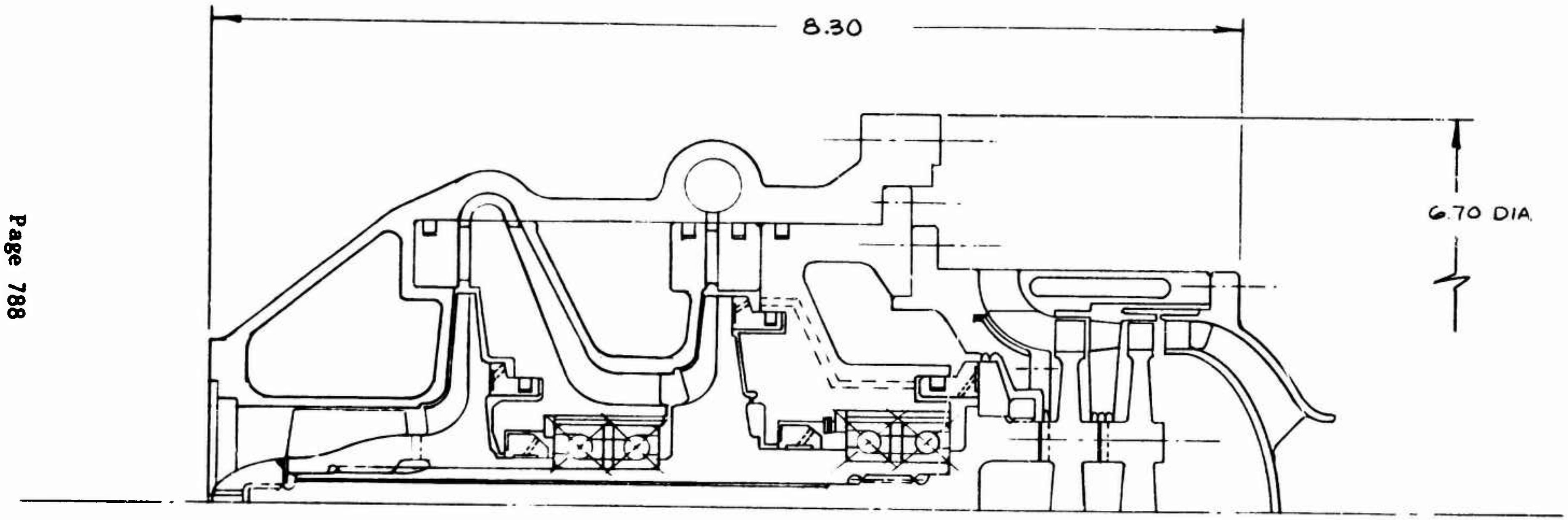


Figure 330. 10K Fuel Turbopump Concept, VI - Back-to-Front Unshrouded Impellers - Bearings between Stages

TABLE CXVI

FUEL TPA CONCEPT EVALUATION

Fuel TPA Concept Number	Description	Advantages	Disadvantages	Rating
I	Back to Back Unshrouded Impellers - Bearing between High Speed Inducer and First Stage Impeller	Off Setting Pump Axial Thrust-First to Second Stage. Leakage of Heated High Pressure - Vapor Entrained LH_2 into Suction of First Stage Impeller Eliminated.	Large Hub Ratio First Stage Impeller. Bearings between Inducer and 1st Stage Impeller preclude shroud driven high speed inducer. Tight axial clearance required. Increase passage length and complexity.	5
II	Back to Back Shrouded Impellers - Bearing between High Speed Inducer and First Stage Impeller	Off Setting Pump Axial Thrust - First to Second Stage. Tight Axial clearance not required.	Large Hub Ratio First Stage Impeller. Bearing between Inducer and 1st Stage Impeller preclude shroud driven high speed inducer. Leakage of Heated High Pressure - Vapor entrained LH_2 into suction of first stage. Increased passage length and complexity.	6
III	Front to Back Shrouded Impellers - Bearing between High Speed Inducer and First Stage Impeller	Tight Axial clearance not required.	Large Hub Ratio First Stage Impeller. Bearing between Inducer and 1st Stage Impeller preclude shroud driven high speed inducer. Leakage of Heated High Pressure - Vapor entrained LH_2 into suction of first stage.	4
IV	Front to Back Unshrouded Impellers - Bearing between High Speed Inducer and First Stage Impeller	Leakage of Heated High Pressure - Vapor Entrained LH_2 into suction of first stage impeller eliminated	Large Hub Ratio First Stage Impeller. Bearing between Inducer and 1st Stage Impeller precludes shroud driven high speed inducer. Tight axial clearance required.	3
V	Front to Back Shrouded Impellers. Bearing between Stages	Tight Axial clearance not required. Shroud driven high speed inducer feasible. Small Hub Ratio First Stage Impeller	Leakage of Heated High Pressure - Vapor Entrained LH_2 into suction of first stage.	1 (selected concept)
VI	Front to Back Unshrouded Impellers - Bearing between Stages	Leakage of Heated High Pressure - Vapor Entrained LH_2 into Suction of First Stage Impeller eliminated. Small Hub Ratio First Stage Impeller	Unshrouded impeller precludes shroud driven high speed inducer. Tight axial clearance required.	2

(d) Oxidizer TPA Candidate Configurations

Two configurations of the oxidizer turbopump were considered in the oxidizer TPA selection. The two candidates are shown by conceptual sketches in Figures 331 and 332. Concept No. I has a one and one-half main stage and Concept No. II has a single main stage. Concept No. I with a one and one-half stage pump uses the first stage to raise the pressure level of the oxidizer to that required at the injector of the secondary combustor. The one-half stage then receives approximately 15% of the oxidizer flow and increases its pressure level to that required at the injector of the primary combustor. Concept No. II with a single stage pump raises the pressure level of the total oxidizer flow to the high pressure level required at the injector of the primary combustor. The pressure level of the oxidizer that is going to the secondary combustor (85% of the flow) is reduced to the pressure required at the injector of the secondary combustor. This throttling of pressure in the oxidizer circuit which is an internal loss in the propellant feed systems, does not permit a system design with a thrust chamber pressure of 1250 psia. Therefore, Concept No. I which does have the capability of achieving the desired 1250 psia thrust chamber pressure was selected.

(e) Low Speed Pump Drive Candidates

The low speed pump drive candidates considered for the 25K engine TPA were reconsidered for the 10K engine TPA's. Two of the six candidates, the full flow hydraulic turbine and electric motor drive, were rated 1 and 2 in that order and were evaluated in greater depth. Two electric motor drive concepts permitted tank mounting of the boost pumps and with a separate electric motor drive, these pumps would also be used in bleeding the main pumps. The shaft power requirement, 6 for the oxidizer and 3.3 HP for the fuel boost pumps, is sufficiently high that the electrical supply system weight and complexity outweigh the advantages of electric motor drive. Therefore, the full flow hydraulic turbine was selected for the 10K engine turbopumps.

(f) Materials

The materials selected for the 10K engine fuel and oxidizer turbopumps are the same as selected for the 25K engine turbopumps. A complete discussion of the rationale of material selection is included in Section III,B,2,g.

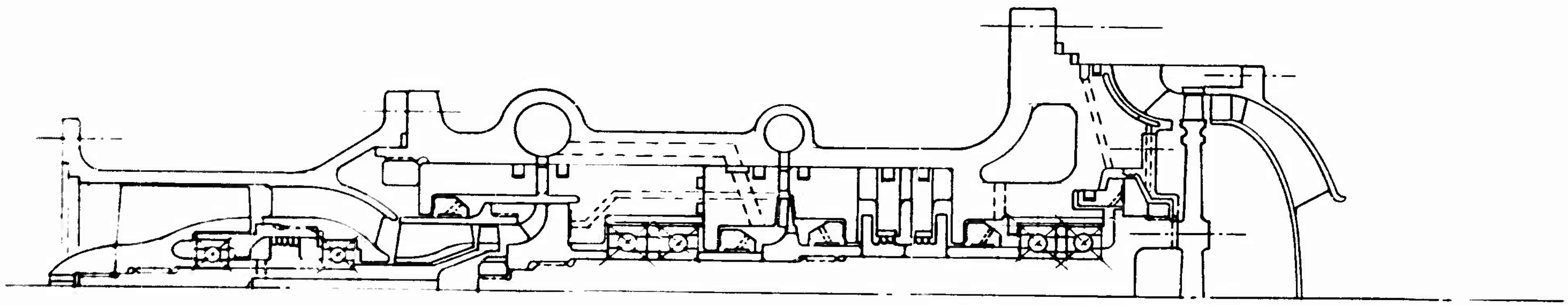


Figure 331. 10K Oxidizer Turbopump Concept, I - One-and-One Half Mainstage

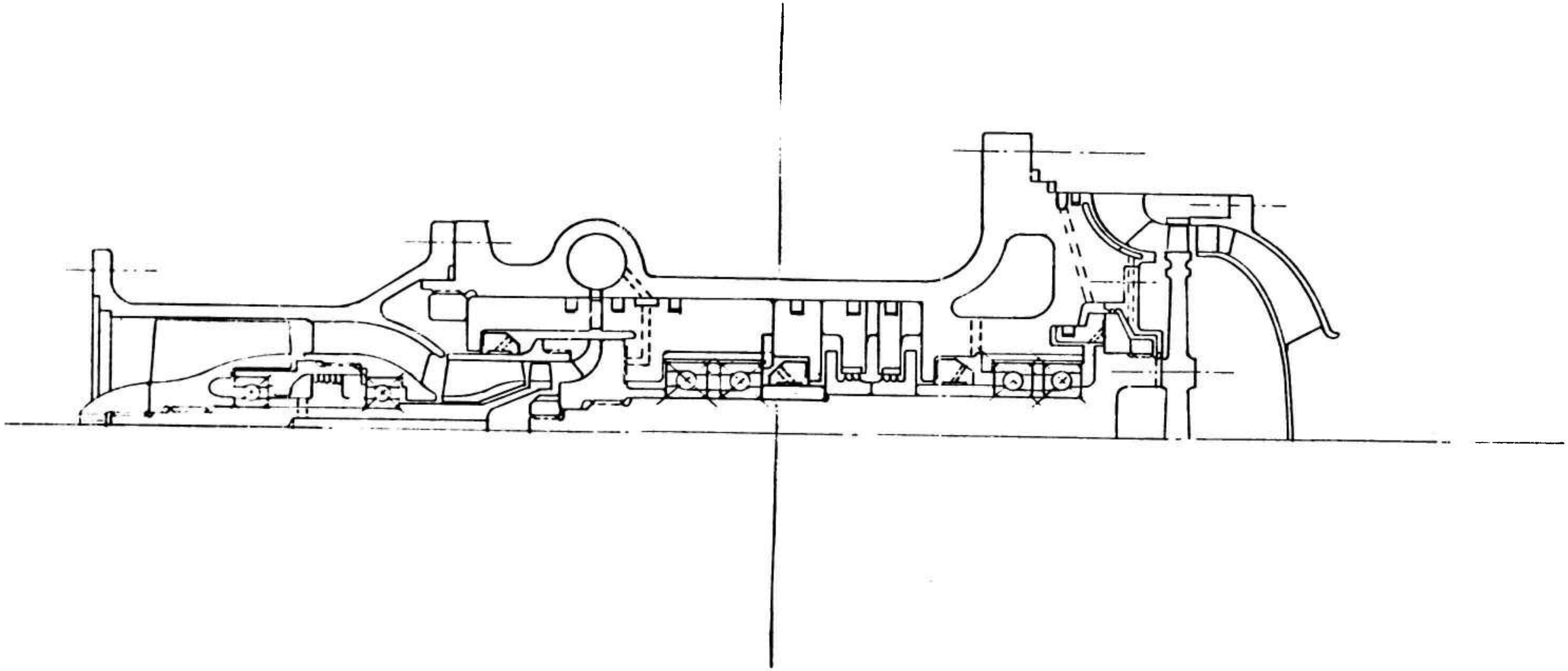


Figure 332. 10K Oxidizer Turbopump Concept, II - Single Mainstage

(3) Description

(a) Fuel Turbopump

1 Assembly Description

The fuel turbopump shown in Figure 333 consists of a high speed main turbopump and a low speed turbopump. The low speed turbopump consists of a low speed inducer and a full flow hydraulic turbine mounted to a common shaft. Propellant lubricated ball bearings support the shaft. The full flow hydraulic turbine is located between the high speed inducer and the first stage pump of the high speed main turbopump. The high speed main turbopump consists of a high speed inducer, two stage centrifugal pump and a two stage turbine mounted to a common shaft. Propellant lubricated ball bearings support the shaft. The design of the fuel turbopump assembly is almost identical to that of the 25K engine with appropriate scaling factors. Consequently the appropriate section relative to the 25K engine constitutes a design description for the 10K engine components as well as the 25K engine components.

The component weight breakdown for the fuel turbopump is included in Table CXVII.

(b) Oxidizer Turbopump

1 Assembly Description

The conceptual design of the main oxidizer turbopump is shown in Figure 334 with a low speed inducer upstream of the main pump which is driven by a " full flow" turbine located between the high speed inducer and first-stage impeller. The main pump is a "stage and a half" design with the first-stage discharging 85% of the flow to the secondary injector and 15% to the half-stage which discharges its flow to the primary injector. The impellers are shrouded to minimize tolerance requirements for tight axial clearances and to provide bias loads for the single acting thrust balancer. The pump is driven by a single-stage partial-admission turbine.

As is the case with the fuel turbopump assembly the design similarity of the 10 and 25K oxidizer turbopump assemblies make it possible to refer to Section III,B,2,g for a description of the 10K design. The component weight breakdown for the oxidizer turbopump is included in Table IV.

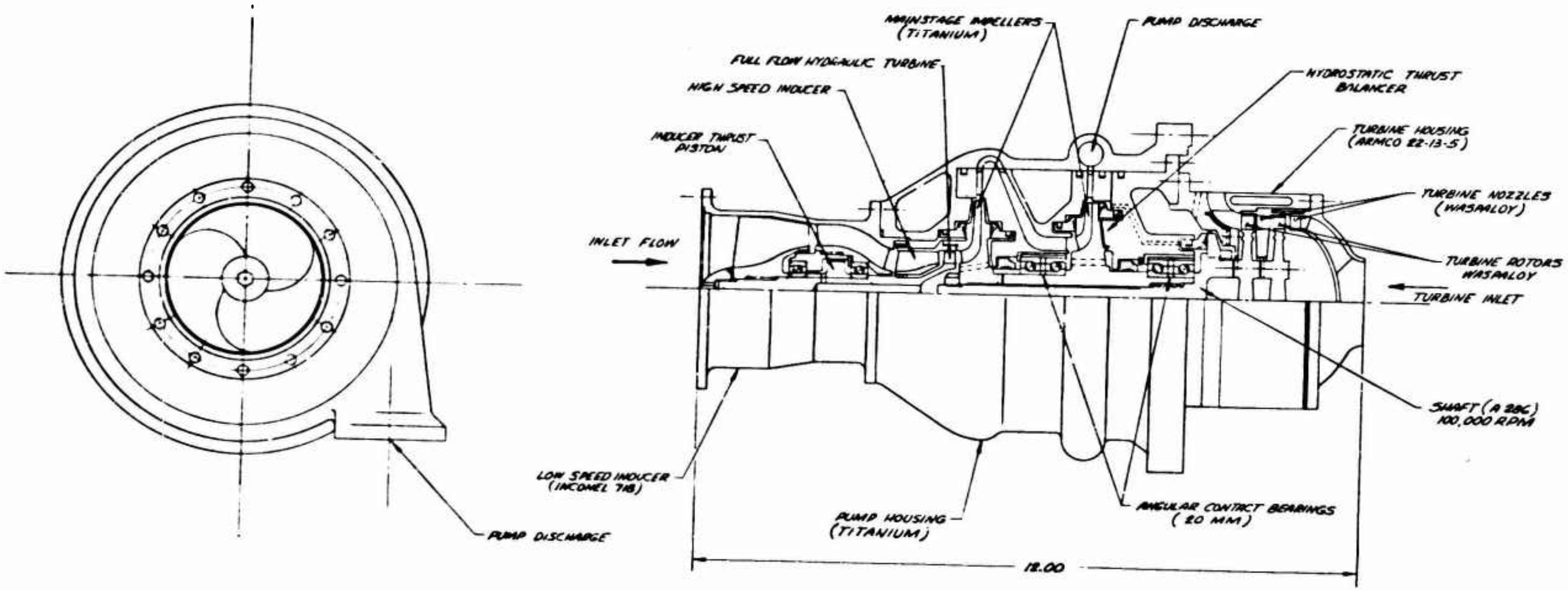


Figure 333. 10K Fuel Turbopump

TABLE CXII

10K OOS TURBOPUMP WEIGHT BREAKDOWN

	Oxidizer	Fuel
Low Speed Turbopump - Pounds		
Impeller	0.17	0.20
Shaft and Bearing Inner Race	0.26	0.28
Hydraulic Turbine	0.10	0.13
Housing and Bearing Outer Race	1.55	1.63
Total	2.08	2.04
Main Turbopump - Pounds		
Inducer	0.18	0.23
Impeller(s) Main	0.20	1.20
Impeller (Half Stage)	0.08	-
Shaft and Rotating Elements	1.85	1.20
First Stage Turbine Rotor	0.75	0.42
Second Stage Turbine Rotor	-	0.42
Pump Housing and Power Transmission Housing	18.31	14.98
Turbine Nozzle Assembly	2.84	2.10
Total	24.21	20.55
Low Speed + Main Turbopumps - Pounds		
Total	26.29	22.59

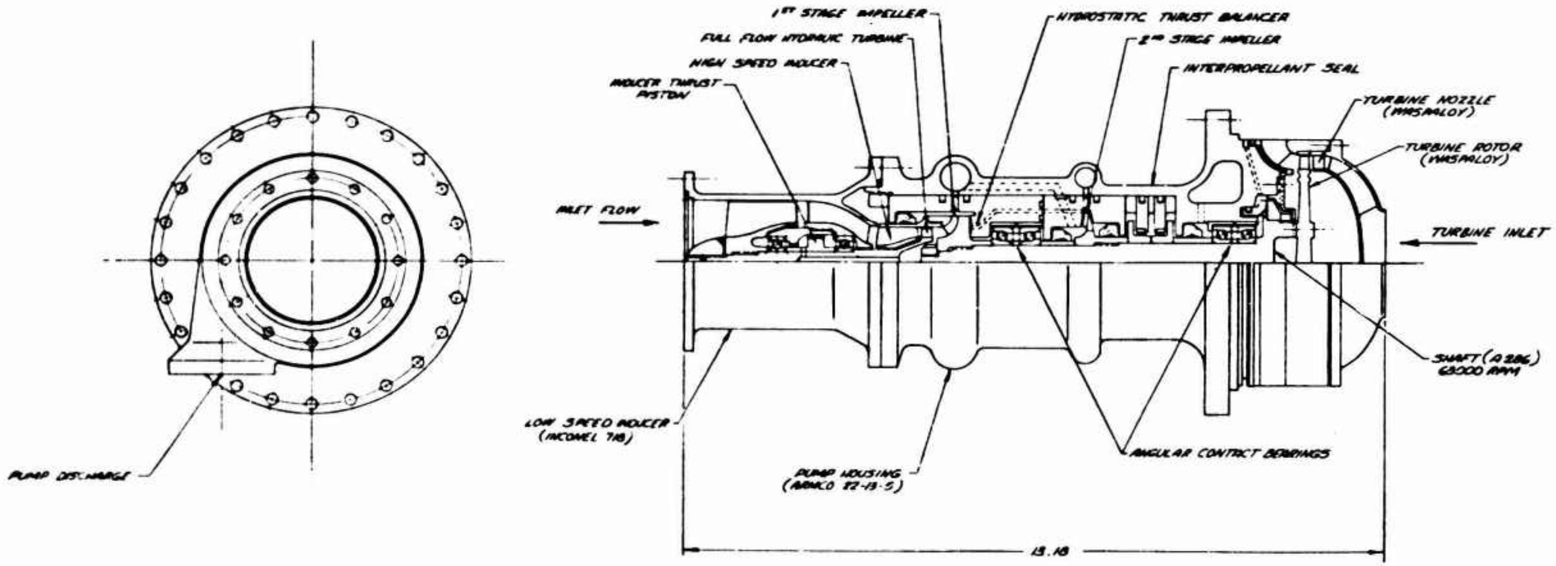


Figure 334. 10K Oxidizer Turbopump

i. Control Valve

The selection process and design features associated with the 10K engine control valves are identical to those of the 25K engine design. Consequently the 10K engine control valves are scaled down versions of those discussed in Section III,B. The flow diameters and weights of the 10K engine control valves are as follows:

	Diameter, in.	Weight, lb
Fuel Pump Discharge Valve	0.75	5.32
Oxidizer Pump Discharge Valve	0.75	5.75
Preburner Oxidizer Valve	0.375	2.5
Fuel Start Bypass Valve (Turbine Bypass Valve same Config.)	0.75	5.75

Figures 335 through 337 show three of the valve concepts. Conceptual drawings of the fuel start bypass and turbine bypass valves are not included because they will be the same configuration as the oxidizer discharge valve.

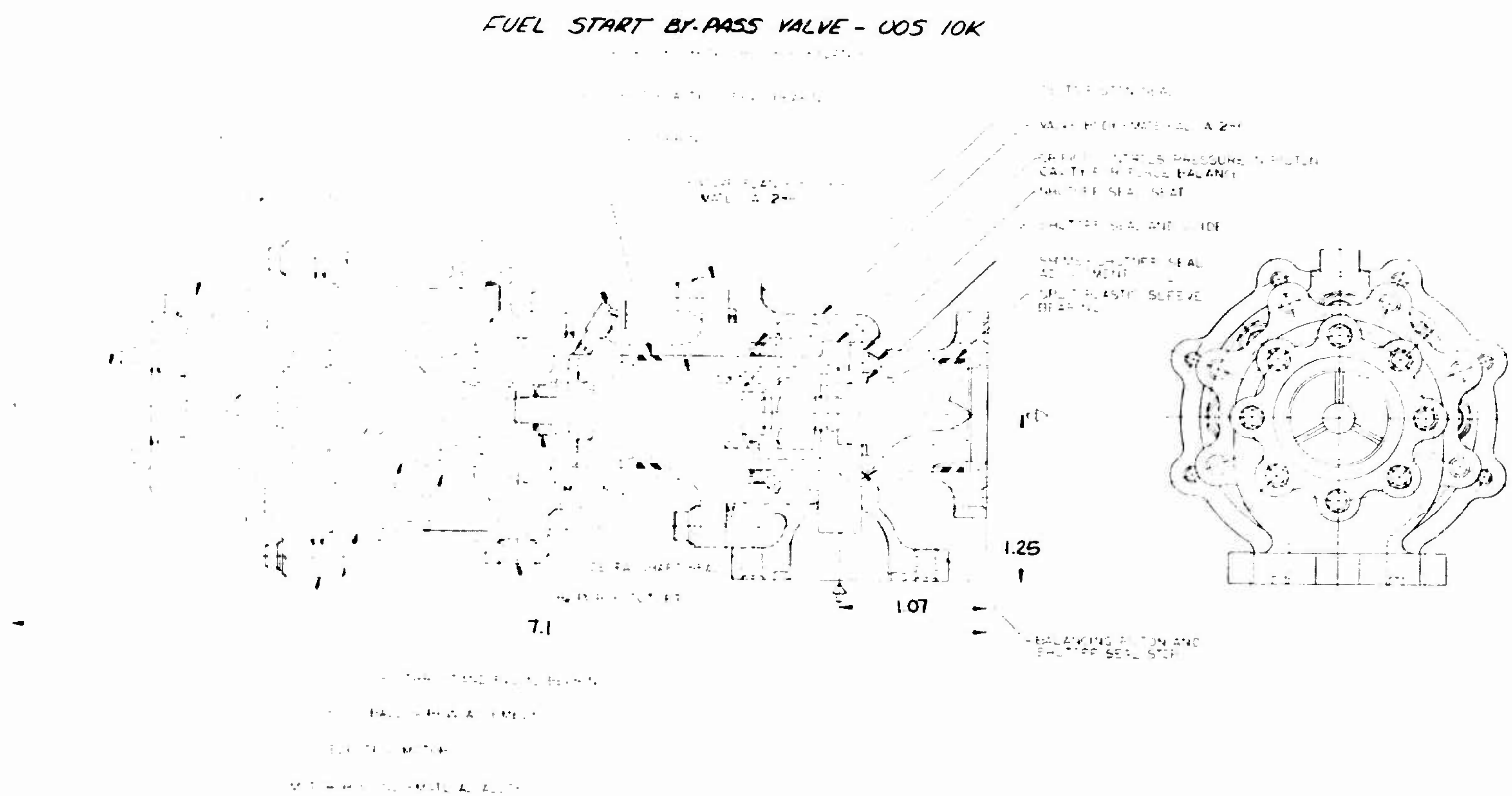


Figure 335. Fuel Pump Discharge Valve

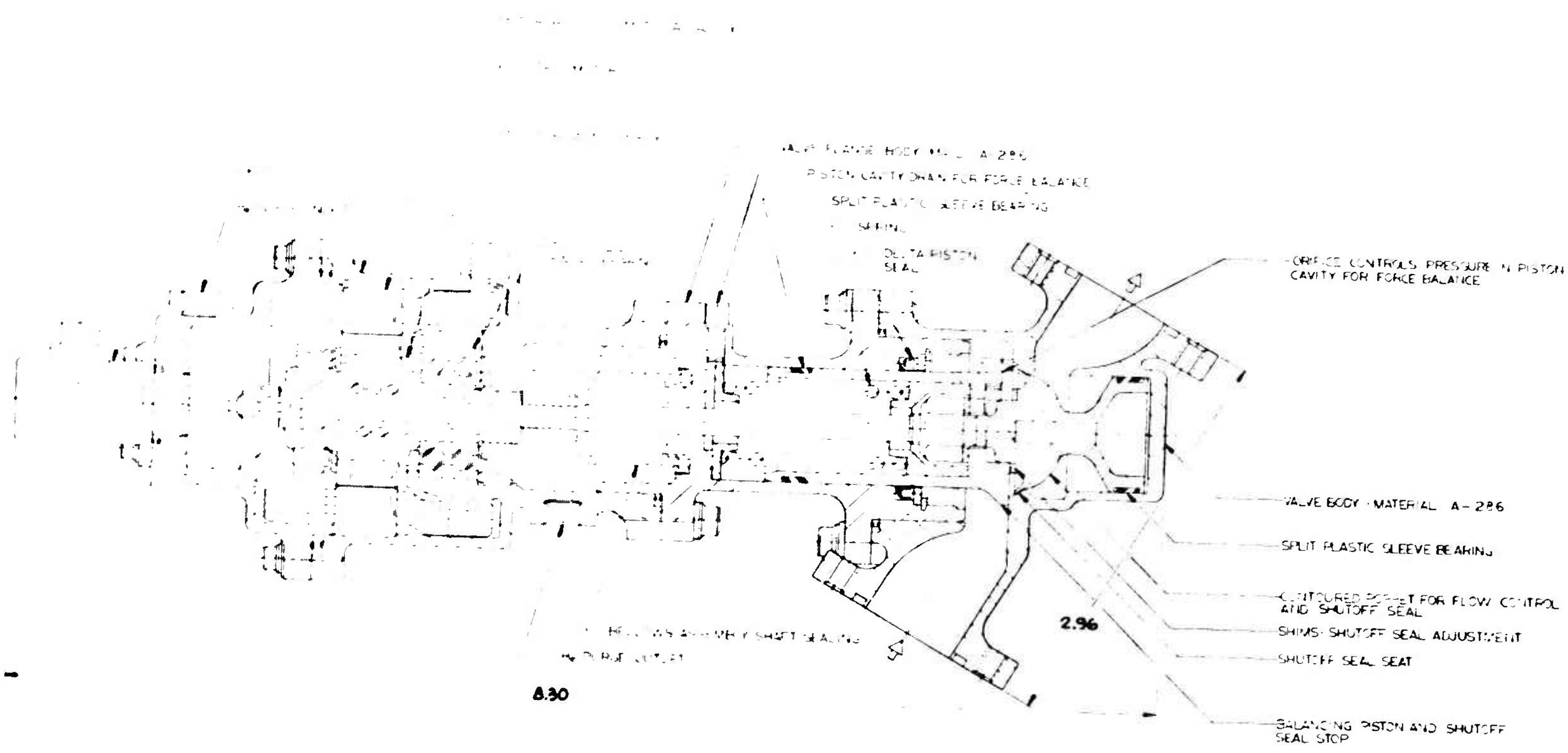


Figure 336. Oxidizer Pump Discharge Valve

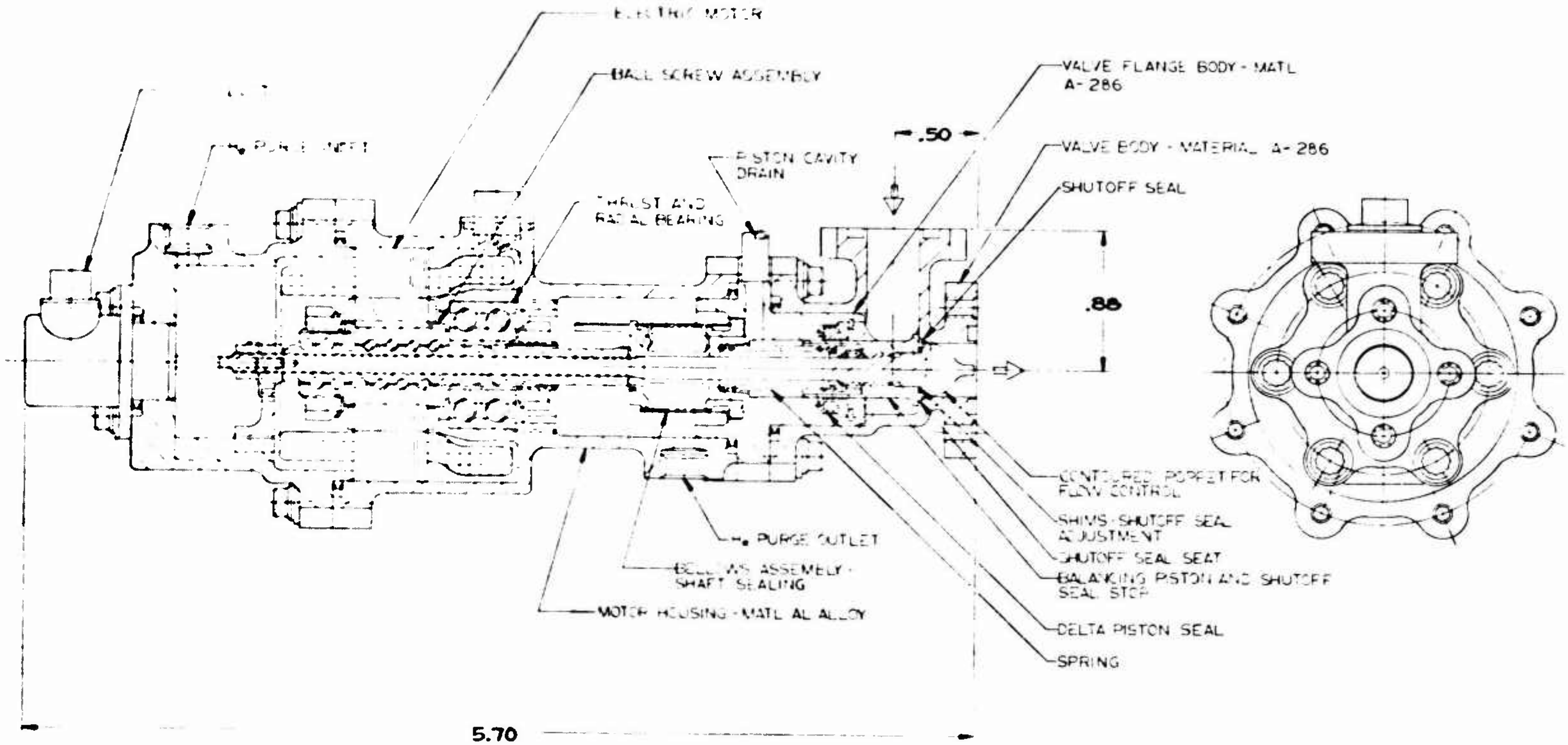


Figure 337. Oxidizer Preburner Valve

F. ENGINE TECHNOLOGY REQUIREMENTS

In course of designing the OOS/Tug propulsion system many technology requirements became evident. Some of these technologies are basic and a function of the engine design and operating requirements, others exist because of uncertainty of the selected approach.

To differentiate in the level of need each new technology required was tagged with a priority, priority 1 being the most needed. The listing shown in Table CXVIII includes all propulsion technologies grouped by components.

TABLE CXIII

OOS RECOMMENDED ENGINE TECHNOLOGY

TURBOPUMP TECHNOLOGY

Technology	Priority	Need	Benefit
Boost Pump Drive	1	• Mechanical Drive or Low Pressure Gas Drive to Eliminate Start Lag	• Elimination of Start Lag • Zero NPSH Restart • Elimination of Tank Repressurization
Rolling Contact Bearings	1	• Limited Experience with Small dia High Speed Bearings • Life, Load and Cycle Data Bearing Retainer Technology	• Rolling Contact Bearings • Data for High Speed Light Weight Pump Design
Hydrostatic Bearings	1	• Hydrostatic Bearing Technology for Transients	• Hydrostatic Bearings Believed not Life Limited
Shaft Seals	1	• Reduction of Interstage Leakage for Small Pumps • Hydrostatic Seal Technology	• Extension of TPA Power Balance Capability. • Increase of Thrust Chamber Pressure

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
Thrust Balance	1	• Small Rolling Contact Bearings must have Light Load for Life Requirements	• Engine Life, Maintenance Cost, Reliability
Hydrogen Cooled Gear	2	• Feasibility of Gear Driven LOX Pump for Small Thrust Engines	• Elimination of Hot Gas Manifolds. Substantial Weight Beneift.
Split Flow LOX Pump	3	• Stage Combustion Cycle Need 1 1/2 Stage LOX Pump	• Simplicity of Pump Design • Reduction of Bearing Span
Interpropellant Seal	4	• Demonstration of Vented Seal	• Elimination of Purge Requirements
		THRUST CHAMBER TECHNOLOGY	
Chamber Life	1	• Thrust Chamber Material Cycle Life Evaluation in Actual Operating Environment and Conditions	• Increase in Thrust Chamber Pressure. Increased Performance in Length Restricted Propulsion System

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
Large Area Ratio Nozzle Testing	1	• Performance Potential of Large Area Ratio Nozzles in LOX/LH_2. • Throttling Performance • Ice Formation Characteristics	• Performance Demonstration. • Engine Area Ratio Optimization. • Engine Weight.
Nozzle Extension	2	• Evaluation of AGCarb Nozzle Extension Transition Flange Configuration	• Engine Weight Reduction. • Ground Testing and Handling Capability.
Separable Nozzle Extension	2	• Definition of Ground Engine Testing for Definition of Transition Area Ratio • Transition Flange Design	• Reduction of Ground Facility Cost • Capability of Using Aspirator for Checkout Instead of Steam Ejectors.
LOX Vaporizer	2	• Autogenous System. • Gas/Gas Injection • Pump Assisted Idle Mode	• Vaporizer Design Information for Throttleable Engines. Subcritical Vaporizer Characteristic Definition.

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
Chamber Length	2	• Injector Concept for Minimum Chamber Length at Max ERE	• Short Engine Length for Max Area Ratio and Performance
Stability	3	• Baffle Cooling Technology. • Acoustic Damper in Main Chamber and Preburner.	• Stable Engine Operation.
Short Nozzle	2	• High Area Ratio in Short Length. • Retractable Nozzle Extension • Force Deflection Nozzle	• High Performance in Length Restricted Propulsion System
		GAS GENERATOR DESIGN	
Injector	1	• Injector for Deep Throttle Capability and Large P.U. Control • Gas/Gas Injector Concept	• Simple Injector Concept • Operational Flexibility
LOX Vaporizer	1	• Gas/Gas Preburner Injector for Operational Capability • LOX Vaporizer Characteristics	• Light Weight Preburner Design • Operational Flexibility • Deep Throttling • Large P.U. Range

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
		ENGINE SYSTEM	
Hot Gas Valves	3	• High Response MR and Thrust Control • Turbine By Pass and Turbine Discharge Valve	• Min Interaction Control • Simplification of Control System
Valve Drive Selection	3	• Selection of Actuator Electric Drive • Linear Actuator Drive for Improved Response Requirements	• Reliability of Control Maintainability
Flow Meters	2	• Prop. Flow Measurement for Feedback Control and Transient Control	• Min. Propellant Outage and Reserve Requirement
Maintenance	3	• Sensor and Instrumentation Minaturization	• Engine Weight Reduction Maintenance Capability

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
Maintenance Philosophy Definition	2	• Definition of Instrumentation Requirements • Engine Controller Definition • Integration of Maintenance in Engine Design	• Maintainable Engine Design • Maintenance Concept Evaluation in Engine Development Phase
Ground Level Maintenance Definition	2	• Ground Checkout and Test Facility and Equipment Definition	• Impact Evaluation on Engine Design
		GIMBAL SYSTEM	
Gimbal Actuator	3	• Selection of Actuator Pneumatic Electric • Engine Supported Linear Self Locking Actuator	• Propulsion Systems Weight Reduction Reliability Interface Simplification

TABLE CXIII (cont.)

Technology	Priority	Need	Benefit
Gimbal Propellant Lines	3	• Capability of Large Gimbal Angle for Engine Out Capability • Low Pressure Drop Gimbal Lines • Life Capability of Gimbal Bellows	• Min. NPSH Requirement • Life Data • Dual Engine for Engine Out Capability
ACS Thrusters	2	• 20 lb to 30 lb LOX/LH_2 Ignition System Valving	• High Performance • Propellant Commonality • Life Demonstration
Feed System	2	• Make Up System Definition • Liquid Main Tank Fed Ullage Gas Compresser	• Mission Flexibility • Tank Prepressurization

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13. ABSTRACT

This report presents the analytical design of propulsion systems utilizing LOX/Hydrogen propellants to be used as the propulsion for the orbit-to-orbit space vehicle of 65,000 lb lift-off weight.

The report contains the evaluation of various engine cycles in the thrust range of 8000 lb to 50,000 lb thrust for performance, weight and envelope culminating in the cycle selection and detail design of a 25,000 lb and 10,000 lb thrust engine. The engine concepts are described in sufficient detail to obtain reliable engine weight, performance, envelope information and methods of engine control. The impact of various engine design requirements were evaluated. The engines are designed to be reusable and capable of throttled engine operation and capable of starting in the idle mode operation.

New technology requirements for meeting the engine design and operating requirements are identified.

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
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25K Thrust Engine Design						
Impact on 25K Engine Design and Performance Resulting from Revised Operating Requirements						
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